

FLIGHT SAFETY SYSTEMS FOR  
ADVANCED MANNED  
SPACE MISSIONS-II

by

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The author currently has no responsibility in the development of flight safety systems for projects Mercury, Gemini, or Apollo. The information presented herein therefore represents the author's personal opinion concerning such systems.

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ABSTRACT

This report presents the design fundamentals and overall philosophy of systems designed to safeguard crews during space missions. Particular emphasis is placed upon the flight regime which includes the rise of the vehicle through the atmosphere. A number of current and future spacecraft configurations are used as illustrative examples of the specific hardware requirements imposed upon such systems by various payloads and mission profiles. The methods of malfunction detection and emergency mode operation of the MERCURY, GEMINI, DYNA SOAR, VOSTOK, and APOLLO spacecraft are presented.

A detailed bibliography, including a number of significant references which are not commonly known is also included.

T. F. Heinsheimer  
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## CHAPTER 1 INTRODUCTION

The rapid development of spacecraft and booster rockets for manned space missions, both in the United States and in the U. S. S. R. , has been paralleled by the development of complex systems designed to safeguard the pilot or crew during these missions. Such systems must protect these astronauts from the effects of onboard malfunctions. For the purposes of the following discussion typical space missions may be conveniently considered as combinations of three sub-missions:

1. flight through the atmosphere
2. extra atmospheric flight
3. operations in the vicinity of an extra terrestrial mass, i. e. the moon or a planet

At present, the most stringent requirements for safety systems are in regard to trans-atmospheric flight. This paper will therefore deal primarily with this phase. The problems involved in the other regimes of flight are discussed briefly below.

Safety during the extra atmospheric portion of a mission (orbital flight or post-escape) is supplemented by including in the design of the spacecraft appropriate subsystem redundancy to effect high reliability, and by a monitoring and display system to warn the crew of onboard malfunctions. During the periods of flight when no large propulsion motors are in operation, extreme speed in the detection and correction of these malfunctions is not critical. As split second action is not required, the crew can play a significant part in such situations, effecting repairs or initiating abort procedures when necessary.

Mission trajectories must be selected to allow abort maneuvers, within the propulsive capabilities of the spacecraft, to return the spacecraft to earth in as short a time as possible. (1) (2) Similarly, landing sites on earth must be selected to allow for

return along such non standard trajectories.

Quiescent (engines off) operations in the third regime, <sup>(3)</sup> the lunar or planetary vicinity, involve similar considerations. When spacecraft engines are in operation however, (as in trajectory correction, orbit change, rendezvous, lunar landing, etc. ) the need for a high speed warning apparatus is more critical. In the event of a spacecraft malfunction, continued operation of powerful propulsion systems in space could have the same catastrophic effect as such operation might have in regard to earth launched rockets. An emergency detection system, capable of initiating appropriate action (partial or complete engine shut down, emergency rocket firing, reprogramming of the trajectory, etc. ) will therefore be required for crew protection.

The design principles of such a detection system are very similar to the system designed to operate during atmospheric flight, though the implementation of emergency action would be quite different. Such implementation would depend on both spacecraft design and the conditions of flight. The most critical consideration would be the availability of emergency propulsion and its optimum utilization.

## CHAPTER 2

### SAFETY SYSTEMS - TRANS - ATMOSPHERIC FLIGHT

Turning now to the case of trans-atmospheric flight, it is clear that this phase encompasses two types of maneuver: the launch phase and the return (re-entry and landing). Safety during the latter is enhanced by redundancy of landing sites in case of loss of maneuvering capability and in some cases by allowing the crew to land either within the spacecraft or by individual parachute. The details of some of these landing mechanisms are discussed in later sections.

The flight regime during which automatic or semi-automatic flight safety systems are most important is during the ascent through the atmosphere. During this phase, violent forces are being exerted on the spacecraft and the booster vehicle. The time interval between normal vehicle operation and the sudden eruption of hundreds of tons of propellant into flaming chaos may be less than a single second. The destructive power of a giant rocket booster can easily be calculated. A handy rule of thumb is to assume an explosive potential equal to one-tenth the vehicle's weight, of TNT. A six million pound C-5 (CRONUS) booster would therefore have the explosive potential of 300 tons of TNT. Clearly, the catastrophic failure of such a rocket presents a serious hazard to the crew unless sufficient warning can be given and appropriate emergency action taken.

Such emergency action would most likely include the termination of rocket propulsion and the firing of emergency escape rockets which would quickly send the spacecraft, or a part of it containing the crew and needed emergency apparatus, to a safe distance from the impending explosion of the booster and its unused propellants.

In order to initiate such an escape sequence, a signal must be given by some form of intelligence which has the capability of properly monitoring missile performance, and the

authority to issue an emergency command. A great deal of reliance must be placed in this decision making unit, for a moment's hesitation in commanding an abort - when one becomes necessary - could negate the chances of crew escape from the area of missile explosion; while an abort command issued without proper justification would terminate a mission which otherwise could have continued.

In order to preclude these undesirable possibilities and to assure the greatest effectiveness of such a decision making system, the system must be able to meet the following requirements:

1. Operate without degradation under the severe conditions associated with the rocket environment: noise, "g" loading, vibration, etc.
2. Bear the principal responsibility for both the safety of a number of human beings and the successful completion of the mission, without deviating in performance from pre-flight expectation.
3. Continuously monitor a large number of data channels, each of which, in turn, monitors a booster or spacecraft parameter critical to proper vehicle performance.
4. Assign to each data channel a pre-determined "range of safety" which is indicative of proper operation of the system monitored by that channel.
5. Initiate a MAYDAY command in case a monitored parameter should deviate from this "range of safety."
6. Add new data channels, delete existing channels, narrow, widen or shift the "range of safety" of a number of channels in a pre-determined,

controlled manner, either as a function of time or in response to commands from other systems.

7. Exhibit a speed of response rapid enough to allow sufficient time for the implementation of emergency procedures prior to a booster explosion.
8. Operate on a "non-interference" basis with all other systems including the crew.
9. Exhibit extremely high reliability in both the ability to recognize and react to dangerous conditions, and in the ability to preclude the issuance of an inadvertant MAYDAY command.
10. Perform all above functions without adding a significant weight penalty to the spacecraft or to the rocket booster.

From these considerations it is clear that a system in which one or more of the crew members were employed to monitor the input data and make the required decisions could not meet these stringent requirements. Therefore, an automatic system is required during this ascent phase. This system would accept, process, and evaluate all data required to maintain proper surveillance of critical vehicle systems.

Such a system would be empowered to command MAYDAY, irrevocably initiating the spacecraft abort sequence. Such action will be required in all cases in which speed of response is most critical to crew safety, as in the case of a serious malfunction occuring during the rise of the booster through the lower portions

of the atmosphere.

There may be situations, however, in which alternate modes of action may be taken by the automatic system, and subsequent action taken by the crew or even by ground control. If, for example, the flight safety system discovered a malfunction, while the booster was in powered flight above the denser portion of the atmosphere, the system would classify the malfunction as either a catastrophic failure or a less serious malfunction, which could become catastrophic if immediate action by the safety system were not taken. In the first case, the safety system would issue a MAYDAY command and the mission would be aborted. In the second case, less drastic action could be taken. An engine cut-off command, which accompanies almost all MAYDAY commands sent by the system, could be sent without an associated MAYDAY command. This would terminate the operation of the rocket engines, but would not cause automatic ejection of the crew compartment. The entire vehicle would then enter a coast trajectory which, depending on the time of failure and the trajectory being followed, could be maintained safely for periods of time ranging from seconds (in case of low altitude) to several hours (as in a parking orbit) without compromising the intended mission. During this time, evaluation of the condition of the vehicle and of the failure sensed by the system could be conducted by the crew and ground personnel on the basis of "real time" telemetry data.

While this analysis is being completed, the flight safety system maintains its monitoring capability. If any sensed parameter should deviate from the limits of safety, or the period of safe coast expire, the system would command MAYDAY. Many of the monitored data channels (i. e. propulsion monitors) would be modified to compensate for the unexpected environment in which the system finds itself, due to the interruption of the thrusting program. Such modification would be automatically implemented by the system when it commands engine cut-off.

When the analysis of data is complete, the crew is advised of the conclusions of the ground personnel and is then free to act on this information. It may choose to abort the mission and return to earth immediately or coast for some time longer and then abort, (in order to land in a more favorable location), or to correct the malfunction and then employ the unused propulsive capability of the rocket.

Such a system could be designed to have multiple capabilities in order to properly protect the crew from virtually all contingencies of vehicle failure. It would exhibit the advantages of speed and high capacity inherent in automatic equipment while maintaining the flexibility made possible by the inclusion of man in some of the decision making processes.

The component parts of such a system are illustrated in Fig. 1. The system is centered about a control unit which accepts all required information and determines the degree of flightworthiness of the vehicle. Depending on the particular missile in question, the control unit may be a single package, or might be divided into a number of sub-units, one placed, for example, on each booster stage. The former approach concentrates all decision making power into a single unit, alleviating a number of problems such as the signals required between sub-units, and the apportionment of control to each as a function of time. The latter approach, however, is economical in weight. As each stage is jettisoned, the logic elements that monitored the sensors associated with that stage are also jettisoned. The monitoring task is then the function of the other sub-units, until the next booster stage is discarded, at which time the monitoring task is again automatically re-assigned. This approach relieves the problem of having to carry a large amount of circuitry aboard an upper stage which is not required during the burning period of that stage. Detailed design studies are required to determine which of these approaches may be the most desirable for each particular application.

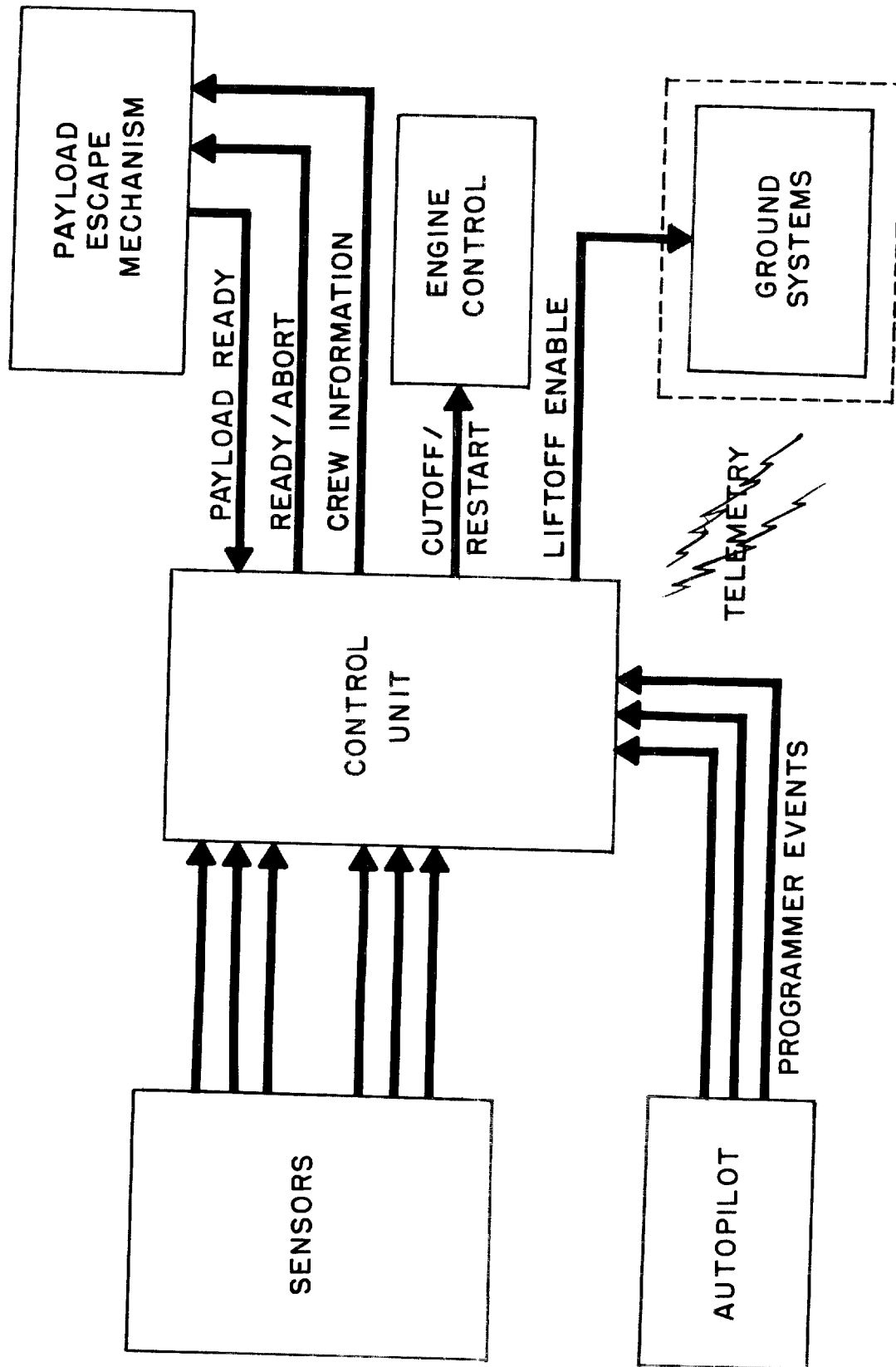


Fig. 1 Flight safety system

Inputs to the control unit are the signals from the monitoring instrumentation and from the programmer. Such information will include data on all missile parameters which are critical to proper missile functioning. At missile lift off, the autopilot programmer begins sending signals, as a function of time, to a number of missile systems. These autopilot commands serve as an indication of "time of flight" and are employed by the control unit as a reference for all "time-based" control unit functions.

The outputs of the control unit may be divided into two broad categories. The first or primary group consists of command signals; the READY/MAYDAY command to the payload, and the CUTOFF/RESTART command to the engine control system. In the case of boosters with controlled release, (in which the missile is held to the launch pad during the first few seconds of engine firing) a third signal may be added to the list. This is the LIFTOFF ENABLE command which is sent from the control unit to the release system on the ground during this hold-down period to enable release. It is sent when all monitored parameters indicate a READY condition.

The secondary set of output signals are those of the non-command type. They are employed by either crew (in the form of panel displays) or ground personnel (after relay through a telemetry link) to evaluate vehicle performance in "real time." These signals may be of vital importance if a malfunction occurs in which CUTOFF is commanded but not MAYDAY, and the vehicle enters a coast phase in which a rapid diagnosis of the malfunction must be made.

To illustrate some of the design problems inherent in such a system, two problem areas will be discussed. The first is the method by which critical parameters are chosen, and the second, an explanation of some of the considerations involved in assigning a "range of safety" to each of these parameters.

A typical space vehicle is comprised of an incredible number of components. The failure of any of thousands of these

parts, particularly within the booster, might quickly lead to a catastrophic failure, while a malfunction of any of thousands more could cause severe degradation of missile performance. Clearly, it is impossible to constantly monitor each wire, conduit, resistor, bearing, pressure bottle, clamp, connector bolt, welded joint, etc., though a failure of any of these could be disastrous. How then is effective monitoring to be established? To simplify this task, the vehicle may be considered as a number of principal systems. These would vary according to the particular configuration under consideration. Considering, for example, a multi-stage liquid propellant rocket, such systems would include an electrical, propulsion, pneumatic and/or hydraulic, and possibly a flight control system. The proper operation of each of these systems, in addition to the maintenance of the structural integrity of the airframe, is an absolute requirement for successful vehicle operation. In this case then, monitoring by the flight safety system may be categorized into parameters which monitor operation of each of these systems.

A list of possible parameters which could be used to monitor each system is shown in Table 1. Such a list is, of course, by no means complete but only illustrates possible solutions to the problem.

In some cases, a single parameter might be sufficient to assure proper monitoring. In other cases, a number of parameters may be required. When the final parameters are selected, consideration must be given to a number of factors. Among these are the following:

1. Availability of a definable range of safety and of a definable danger level.
2. Availability of an appropriate sensor or transducer.
3. Reliability, accuracy, stability, and weight of the sensor.

Table 1

Example of MAYDAY Parameters  
for Liquid Propellant Rocket

System	Parameter	Sensor
Electrical	a-c line voltage	electrical sensing circuits
	d-c line voltage	electrical sensing circuits
	a-c frequency	electrical sensing circuits
	continuity between critical points	electrical sensing circuits
Pneumatic	pressure at selected points in system	pressure transducer
Hydraulic	flow rate at selected points in system	flow meter
Propulsion	pressure at selected points in system	pressure transducer
	flow rate at selected points in system	flow meter
	axial acceleration	accelerometer
Flight Control	transverse acceleration	accelerometer
	attitude	displacement gyro
	missile rotation	rate gyro or angle of attack sensors
Airframe	structural integrity	strain gage
		break wire
Miscellaneous	liquid leak	level sensor
	fire	temperature probe or fire detector

4. Complexity of the circuitry required to monitor sensor output.
5. Complexity of the graphical plot of the "range of safety" vs. time.
6. Length of the warning time between deviation from "range of safety" and catastrophic failure of the booster.

The practicability of each parameter must be carefully evaluated, using these ground rules, when selecting safety system parameters for a particular booster vehicle.

Once the appropriate parameters have been determined, a "range of safety" must be assigned to each. This range may have an upper and lower boundary, or may be bounded on one side only. For an illustrative example, a singly bounded range of safety is shown.

On Fig. 2 a parameter is shown with its associated tolerance. The value of the parameter will remain within the tolerance band, as long as all components upon which it depends operate properly. If this value should reach the "red-line" or danger value shown, a catastrophic condition is imminent. This red-line is established through studies of past flights, of flight simulations, or by other technical considerations. Clearly, the MAYDAY and CUTOFF levels must lie between the tolerance band and the disaster limit. For this illustrative example, the CUTOFF level has been deleted and three possible MAYDAY or ABORT levels (A, B, and C) have been added. The determination of the proper CUTOFF level will follow ground rules parallel to those used in the selection of the MAYDAY level.

An optimum MAYDAY level must be chosen between the tolerance band and the danger level. It must be high enough so that sufficient time is available between the payload escape and booster destruction to assure adequate payload separation. This requirement eliminates C which is too close to the danger level. On the other hand, the MAYDAY level must not be so high that a non-catastrophic deviation from the tolerance band would initiate

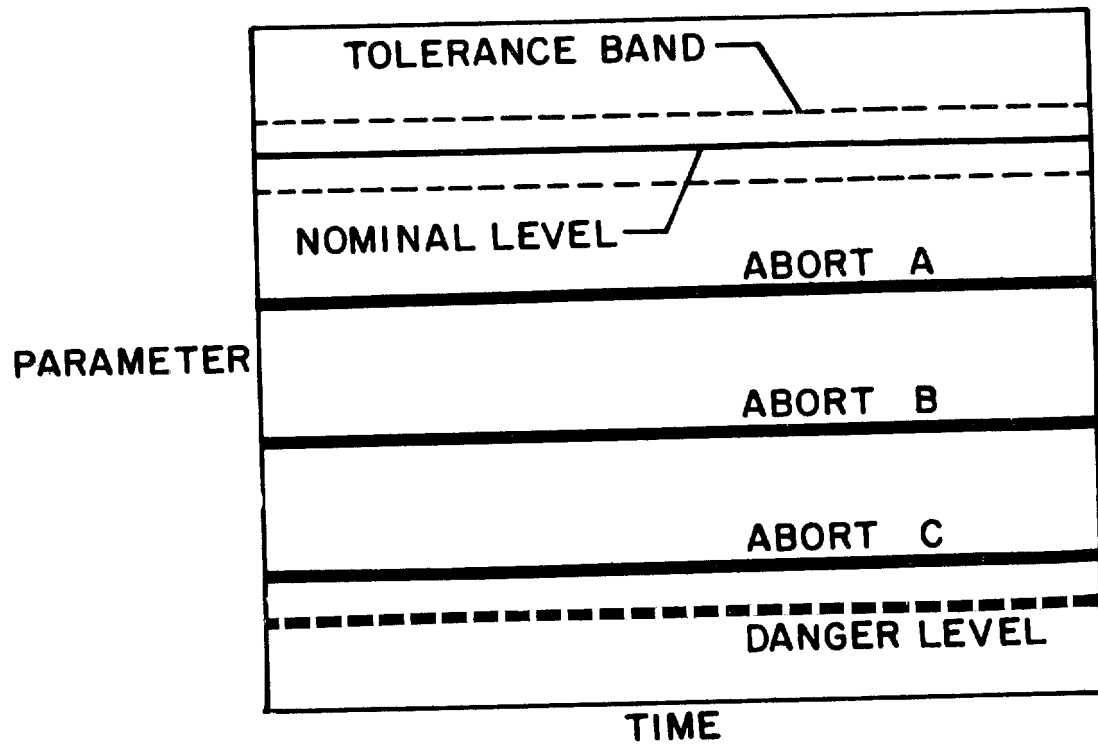


Fig. 2 Abort parameter vs. time (no variations in levels)

payload abort. This requirement eliminates A.

An intermediate level, B, is therefore required. Once this level has been assigned, a tolerance band must be established for the MAYDAY level. This band must be both narrow enough to assure proper operation and wide enough to make practicable the engineering design and the pre-flight testing of the system.

In Fig. 3 an additional factor is introduced. This is the variation of both the danger level and the normal operating level of the parameter under consideration. The requirement that the MAYDAY level become a varying function of time must then be included. Figure 4 demonstrates still another factor, the possibility of non-hazardous transients, which exceed the danger level. If the time period during which the parameter is past the danger level is short enough, the condition is not dangerous. The safety system has a quick enough reaction time to abort the mission on the basis of such transients, if corrective action is not taken. An attenuating filter must therefore be incorporated into the system to smooth these transients. This filter might be an electrical type, acting on the output of the sensor, or might be mechanical, affecting either the sensor's internal characteristics (i. e. damping oil) or the input excitation to the sensor. The degree of filtering incorporated must be sufficient to preclude the possibility of a spurious MAYDAY command being generated by a non-hazardous transient. However, the addition of attenuating filters must be regarded with some caution due to an inherent characteristic of such filters; its time constant. As the attenuating power of a filter is increased, its associated time constant increases also, leading to delays in system response. Excessive delays in the sensing of a legitimate MAYDAY condition, due to the sluggishness of such a filter could cause failure of the payload to escape in time to assure crew safety. Such attenuating filters must be incorporated most carefully, keeping the above considerations in mind.

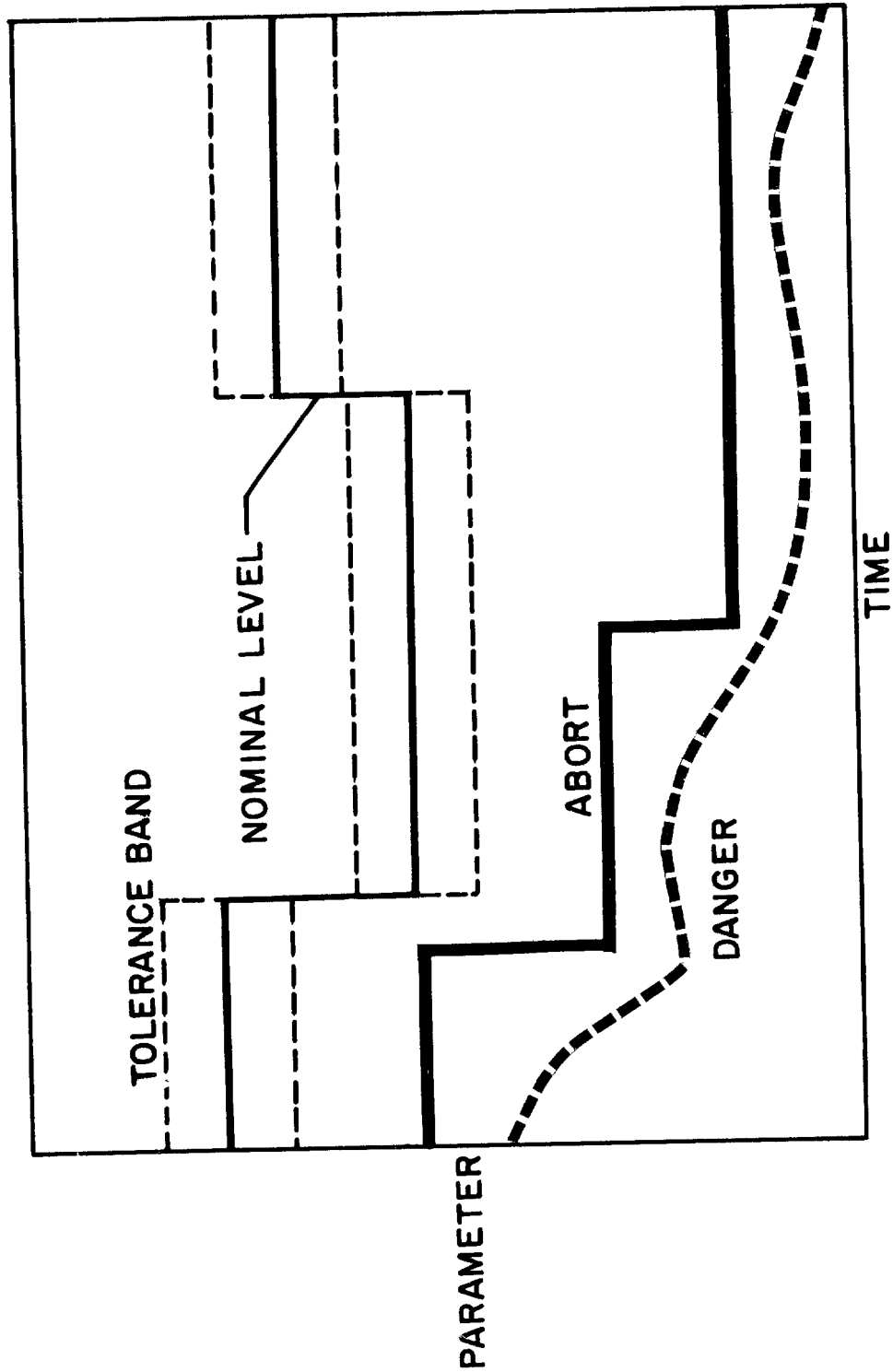


Fig. 3 Abort parameter vs. time (levels changing during flight)

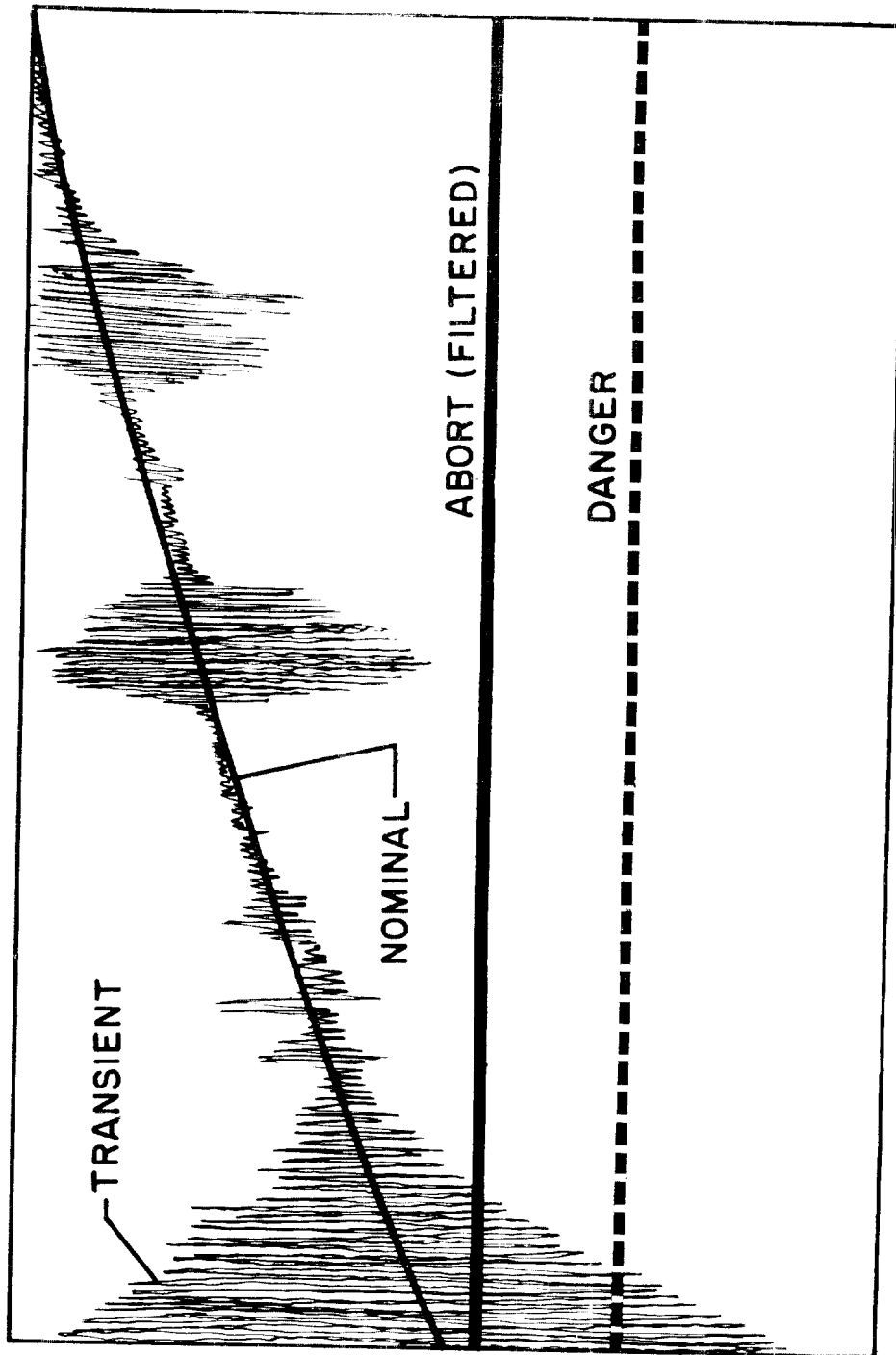
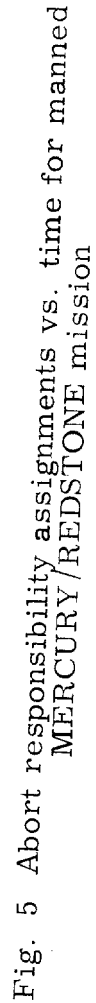


Fig. 4 Abort parameter vs. time (transients extending below danger level)

It should be kept in mind that an abort command could be generated by sources other than the automatic system. In most cases, the responsibility is assigned to the pilot and to predesignated ground based individuals. The launch Test Conductor is customarily assigned an abort responsibility from the time the crew is on board and all pyrotechnics are armed until the vehicle has left the launch area. The Range Safety Officer must monitor the flight trajectory during boost phase and initiate an abort in case the flight path deviates such that it presents a hazard to life and property on the ground. The automatic system must be so designed that upon receipt of a range safety abort signal, it will promptly effect the removal of the spacecraft from the vicinity of the booster and then arm the Range Safety Officer's destruct capability. In this way the system prevents the destruct command from reaching the explosive charge until the crew has been safely removed from the area of the impending explosion. Still another source of abort signals may be the Flight Director, who is in charge of monitoring the flight of the spacecraft during all phases of the mission. The division of abort responsibility during boost phase is shown in Fig. 5 for a typical mission. The figure describes these functions for the MERCURY/REDSTONE mission, the first American vehicle to be adapted for manned space flight. (See Fig. 6.)



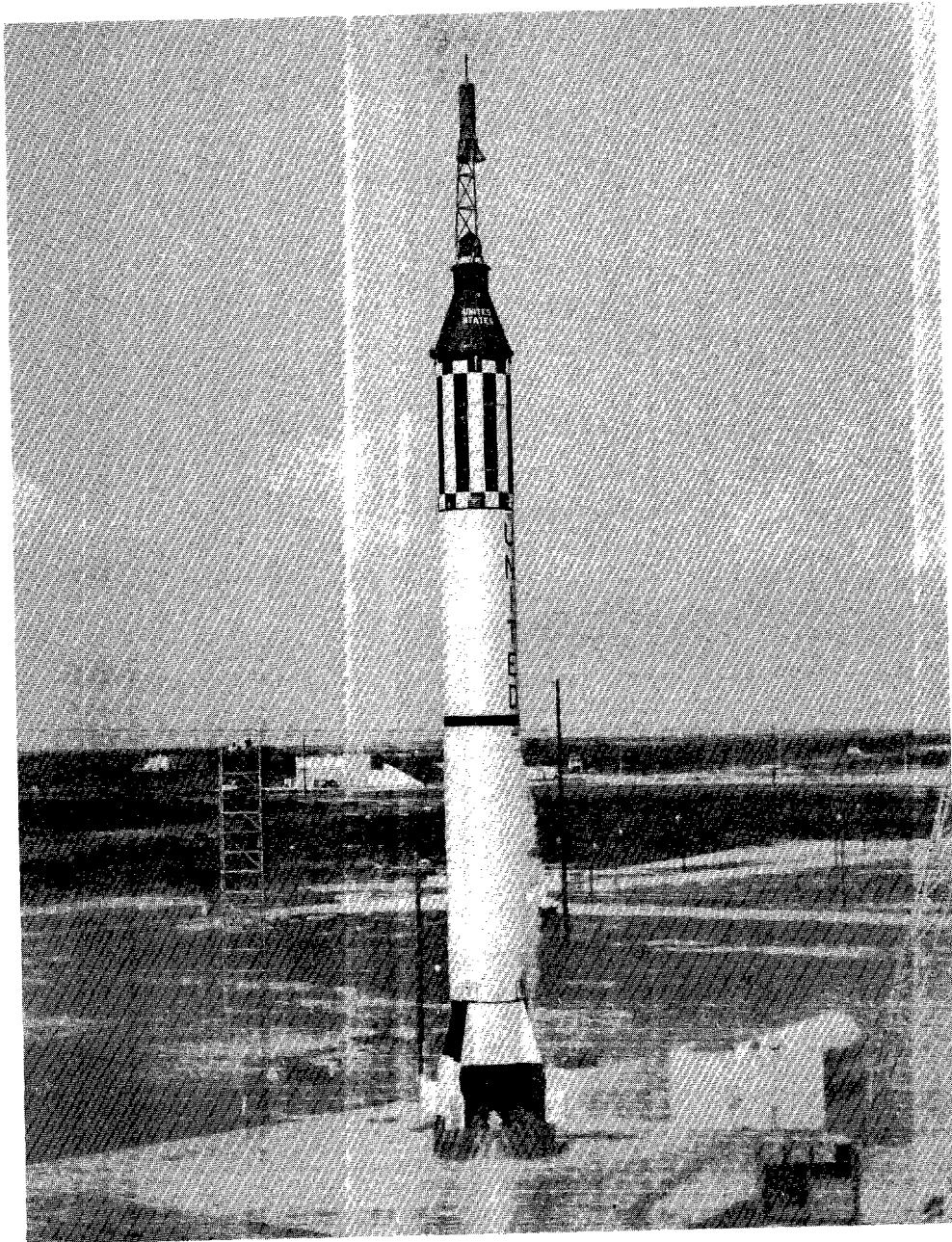


Fig. 6 MERCURY/REDSTONE launch (America's first man in space)

### CHAPTER 3 SPECIFIC FLIGHT CONFIGURATIONS

#### MERCURY Missions

##### A. Implementation of MERCURY Abort

Upon receipt by the MERCURY spacecraft of an abort command during booster phase, a sequence is initiated to free the capsule from the booster. In the case of the REDSTONE flights, the escape tower rocket (52,000 lb thrust for 1 sec) is used, as it is in position (Fig. 7) until normal termination of booster thrusting. <sup>(4)</sup> In the case of ATLAS flights, however, the escape tower is used only until 20 sec after staging. At that time it is jettisoned, and an abort is implemented by firing of the posigrade rockets located under the spacecraft. Jettison of the high impulse escape tower is effected by firing its motors after its separation from the spacecraft. This is to remove the unnecessary weight penalty it imposes once the spacecraft is above the region of significant aerodynamic pressure. After that time most of the hazardous rocket propellant has been burned; the power requirement to remove the capsule from danger in case of an abort has been sufficiently reduced to be satisfied by emergency firing of the much smaller posigrade system consisting of propellant motors, each having 400 lb thrust for 1 sec.

In case of a tower assisted abort, the capsule is allowed to coast to the apogee of its flight path at which time a small tower jettison rocket, (800 lb for 1.5 sec), located at the top of the tower, is fired. This frees the tower from the spacecraft, after which the capsule is returned to earth by parachute. An early test of this procedure is shown in Fig. 8. If an abort should be required during the flight regime in which the tower is no longer present, the posigrade rockets would effect capsule

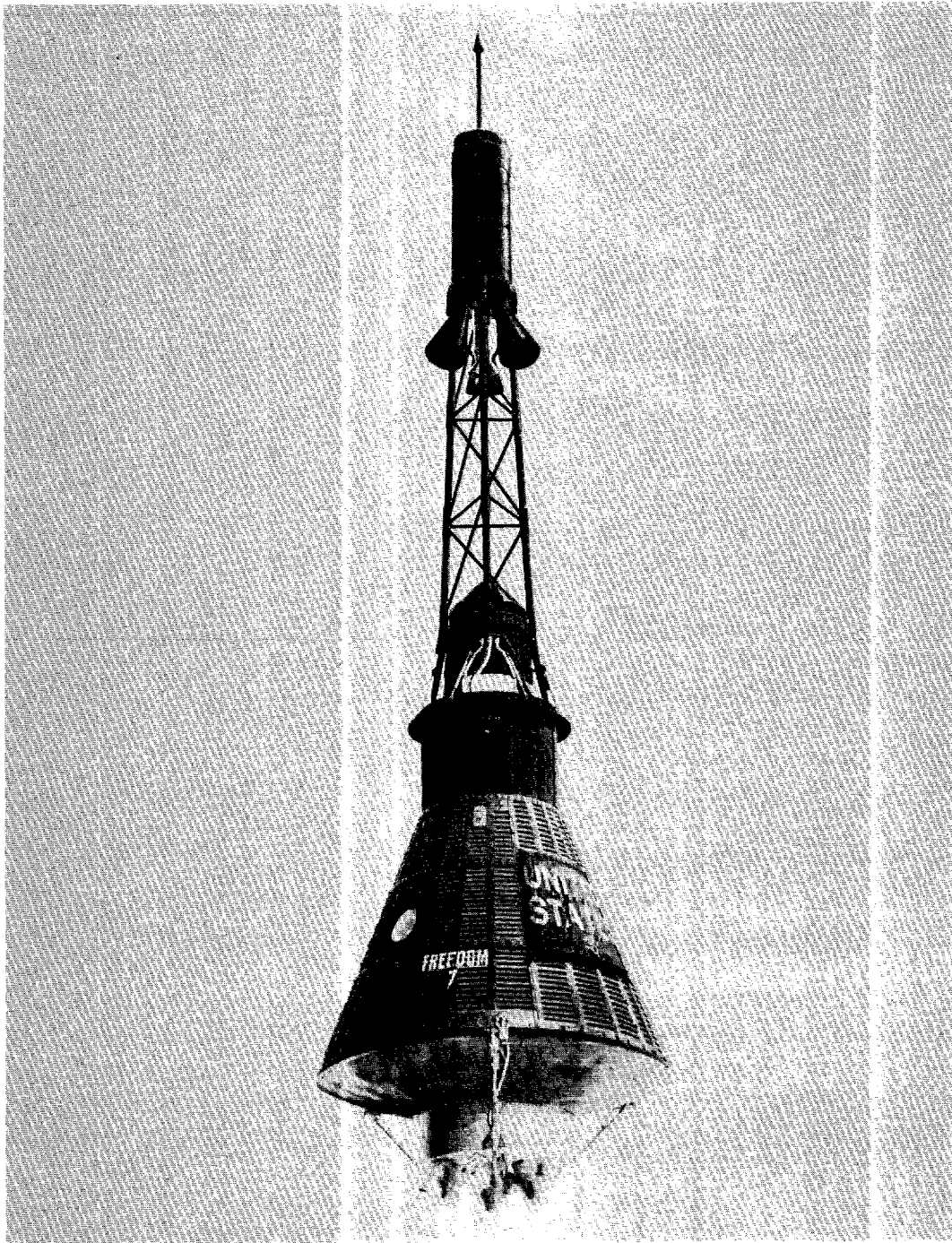


Fig. 7 MERCURY capsule showing escape tower and posigrade rocket configurations

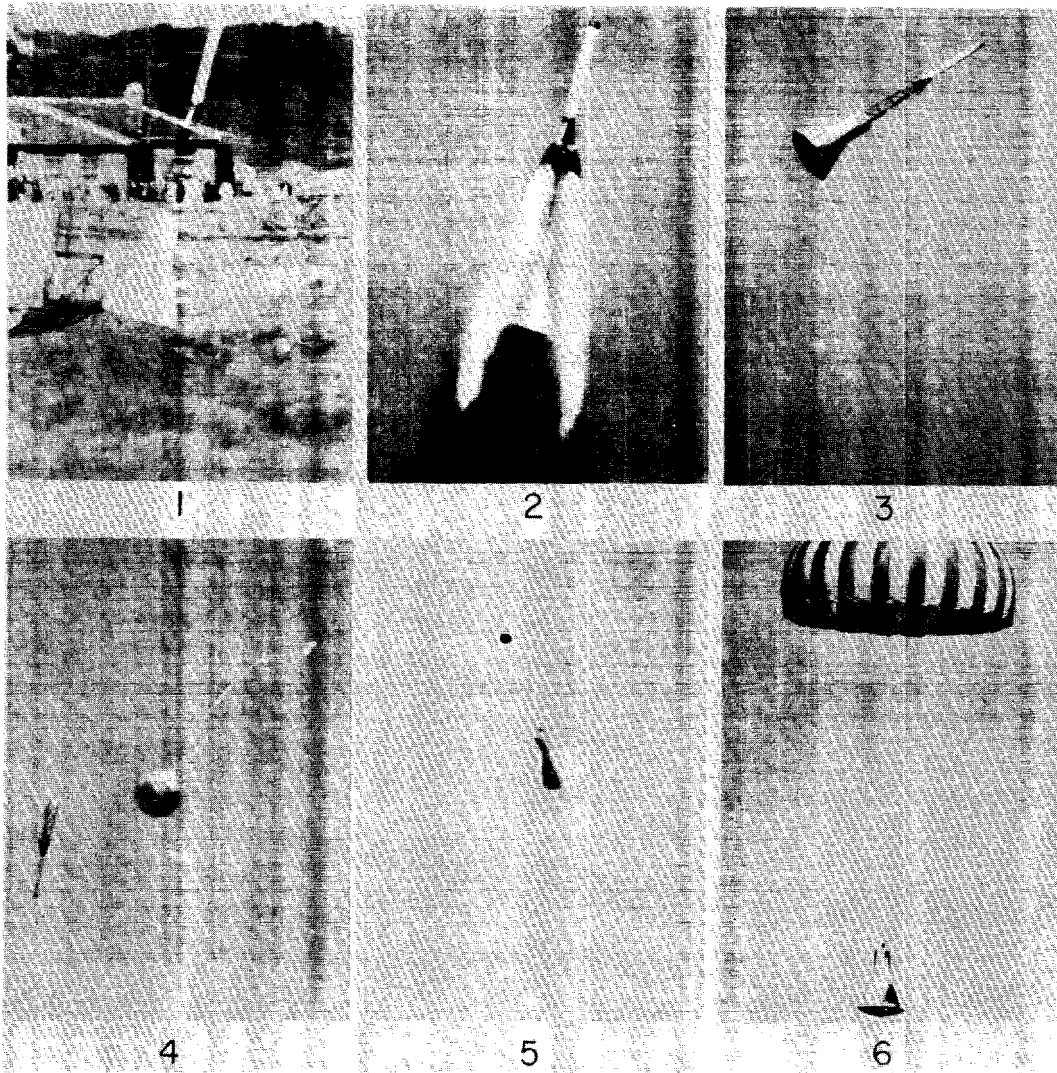


Fig. 8 Early capsule escape tower test

escape, and a ground based computer would determine the requirements for retrorocket firing to assure the proper re-entry conditions, and an appropriate landing site. Re-entry and landing are then completed in the normal manner.

The selection of escape mechanism parameters (thrust, thrust build up and decay, durations, etc.) is the result of many years of exhaustive research and testing. Although the details of these efforts are beyond the scope of this work, applicable references have been included for the convenience of interested readers. References (5) through (15) will be especially helpful in this regard.

#### B. MERCURY/REDSTONE Abort Sensing

The automatic system flown aboard this MERCURY/REDSTONE booster was known as the AUTOMATIC INFLIGHT ABORT SENSING SYSTEM. A block diagram (Fig. 9) shows the relative simplicity of the system, permitted by the greatly reduced complexity of the REDSTONE booster with respect to later vehicles. No control unit was employed in this design, all signals being tied together in an "abort bus" which was routed directly to the spacecraft. Loss of the 28 v on this bus after lift off, initiated the abort maneuver. An automatic engine shut down command accompanies any abort generated at times later than lift off plus 30 sec. Parameters monitored by the system were: pitch, yaw and roll positions, pitch and yaw rates, d-c voltage, and the chamber pressure within the REDSTONE engine.

#### C. MERCURY/ATLAS Abort Sensing

During the early manned flights of the MERCURY/REDSTONE booster, another vehicle was undergoing final testing, prior to its assignment to a manned mission. This was the MERCURY/ATLAS, a modified series D ATLAS, a missile with over one hundred test flights. (See Fig. 10) The MERCURY/ATLAS stands 95' 4" tall, weighing over 260,000 lb at liftoff. 160,000 lb of liquid oxygen and 73,000 lb of fuel are pumped into the five

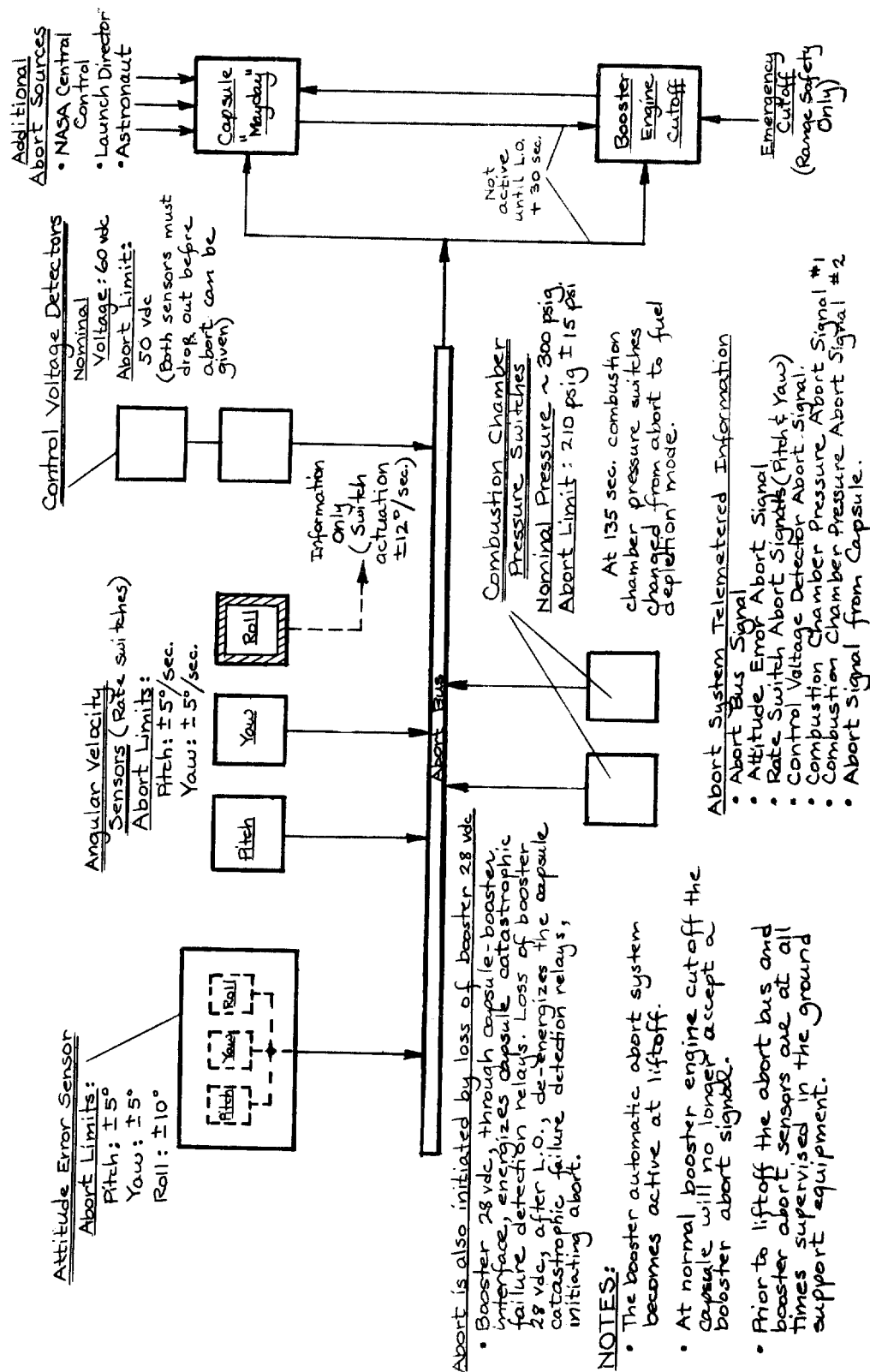


Fig. 9 Block diagram of MERCURY/REDSTONE automatic in-flight abort sensing system

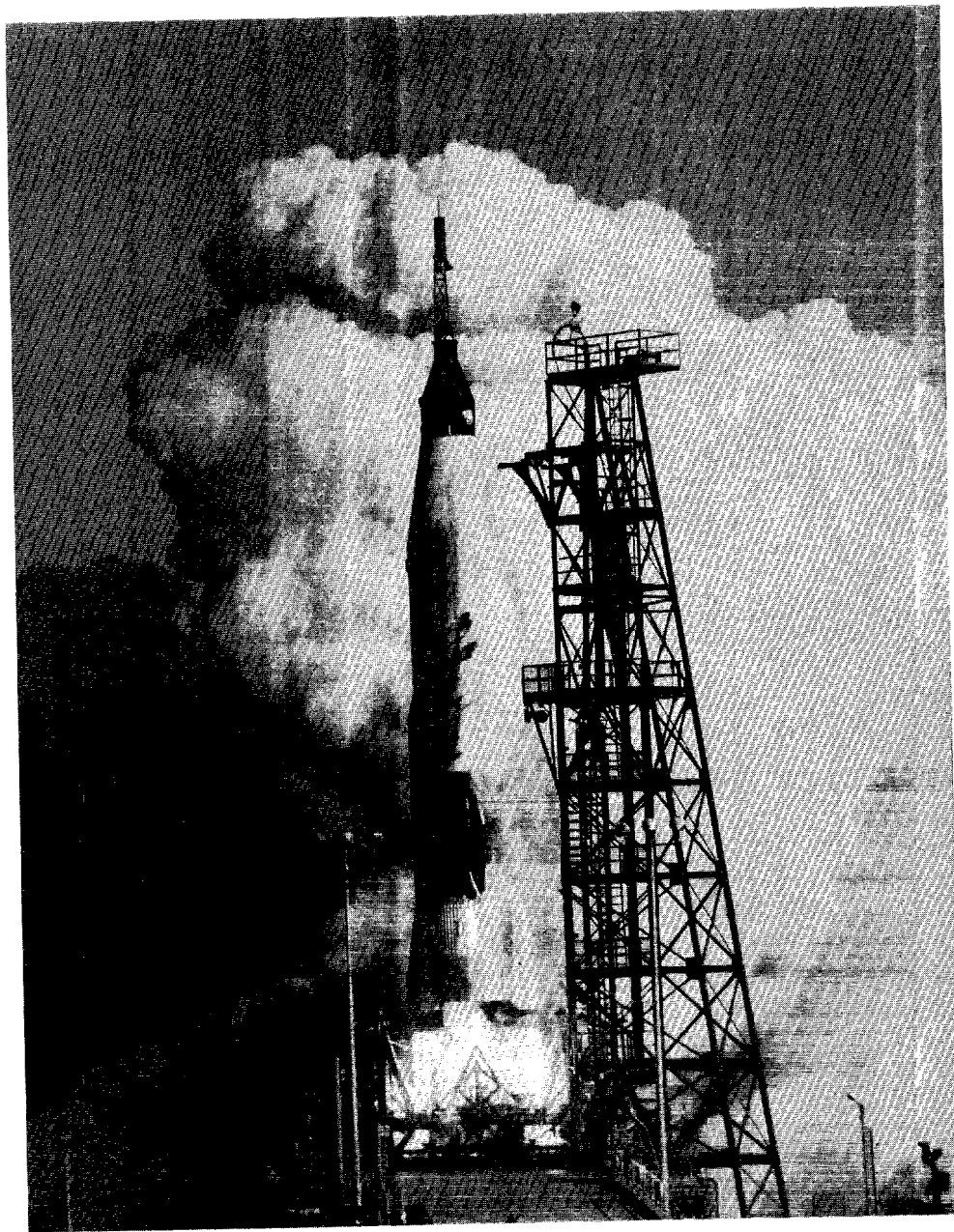


Fig. 10 MERCURY/ATLAS launch

engines, two large boosters of approximately 150,000 lb thrust each, one sustainer of 60,000 lb, and two small stabilizing verniers of 1,000 lb thrust each. <sup>(18)</sup> <sup>(19)</sup> A block diagram of this configuration and the areas monitored by the flight safety system are shown in Fig. 11.

In the case of the MERCURY/ATLAS, this safety system has been named ASIS, the ABORT SENSING AND IMPLEMENTATION SYSTEM. <sup>(20)</sup> <sup>(21)</sup> <sup>(22)</sup> The system monitors the performance of the booster through three basic types of sensors: pressure switches, rate gyros, and electrical circuit components. Information is fed from the appropriate sensors to the control unit, as shown on Fig. 12, within which, as in the model system described earlier, all decisions are made. These sensor inputs are as follows:

- a. Voltages from six rate gyros, two each sensitive to rotation about each of the three missile axes.

These voltages are proportional to the rate of missile rotation. One set of gyros, that which the flight control system uses for missile control, is located in the forward area of the missile, while the redundant set is in the main equipment pod near the control unit itself.

These signals are monitored within the control unit after passing through low pass filters designed to attenuate the effects of non hazardous rate transients as well as bending mode and fuel slosh oscillations.

- b. Signals from two hydraulic pressure sensors.

These sensors, located in the thrust section, near the sustainer engine, monitor performance of the hydraulic system which is required for sustainer engine gimbaling, and sustainer engine shut down. As with all pressure sensors, these units are installed in a redundant manner.

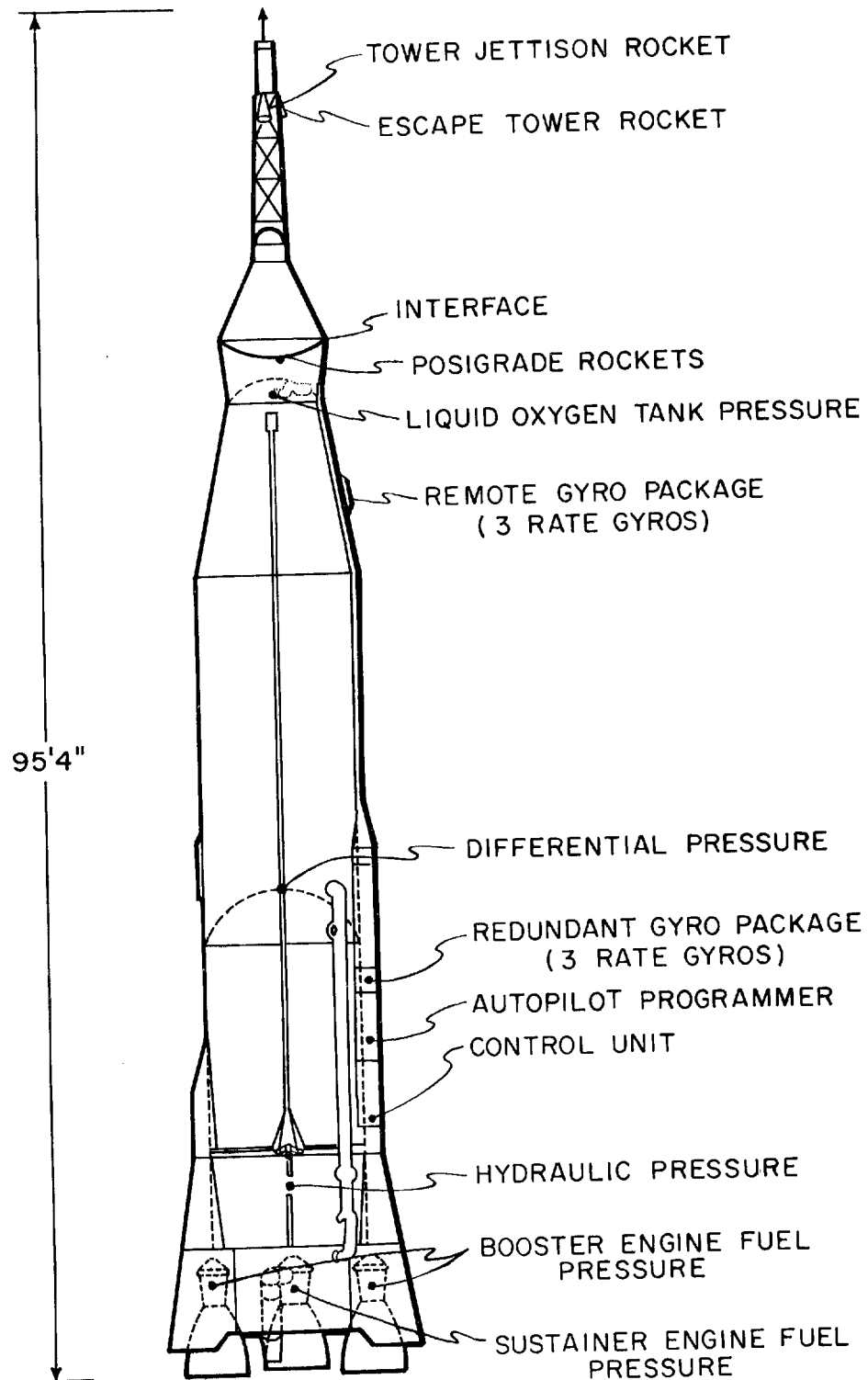


Fig. 11 MERCURY/ATLAS functional diagram

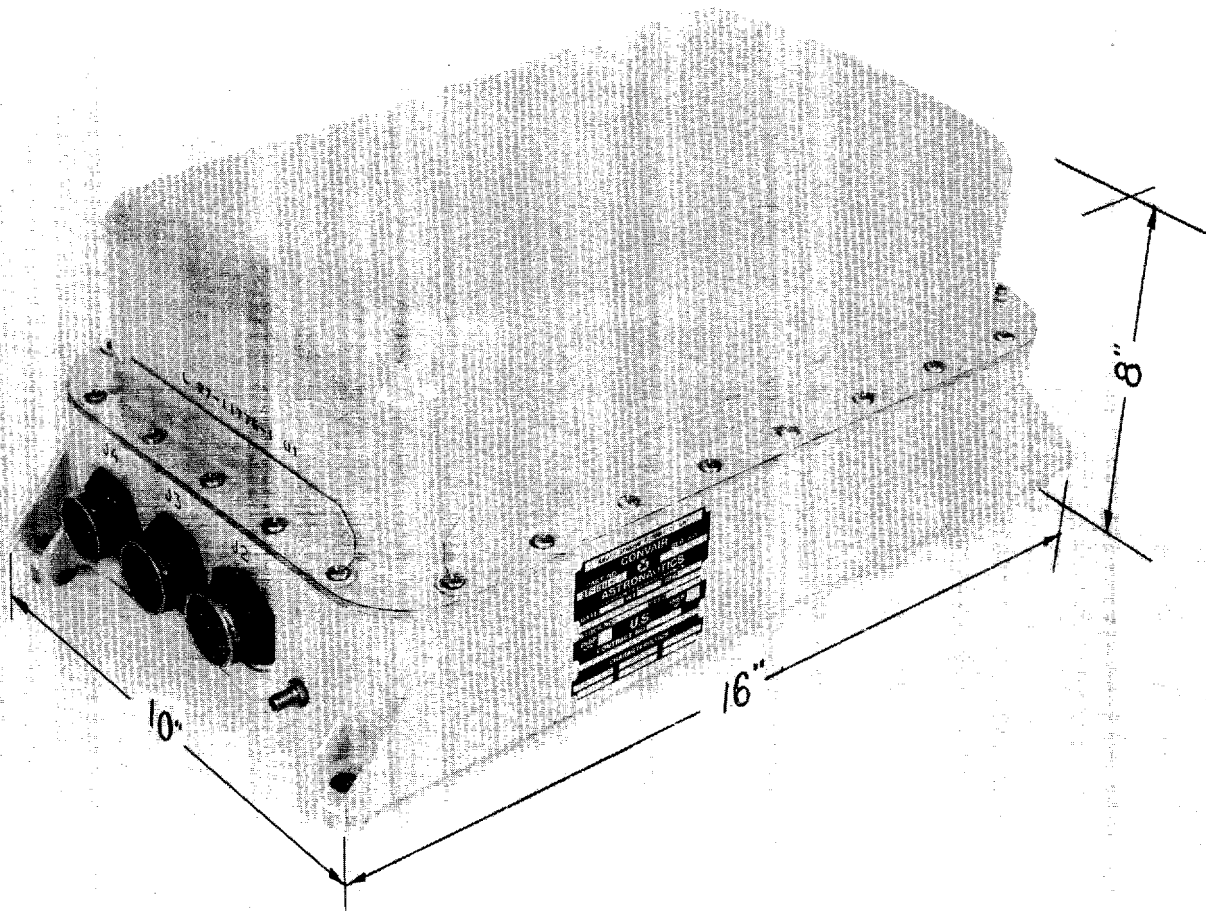


Fig. 12 ASIS control unit  
(unit weighs 34 pounds)

They are constructed by modification of analog pressure transducers to give switch operation. Such modification yields a highly reliable unit as demonstrated by many earlier flights of the unmodified configuration. When the pressure falls to a certain preset limit, the sensor output signal falls to zero.

This zero output mode, indicative of a serious drop in hydraulic pressure, is also the output generated in case of a switch failure, as demonstrated in exhaustive failure mode analyses. Therefore, the control unit is designed to recognize an abort situation only if both switches indicate it. Such applications of redundancy are repeated throughout the ASIS, yielding a high level of fail-safe operation.

c. Signals from six engine pressure switches

As above, two of these switches sense pressure within each of the three principal engines, yielding redundant information about the operation of each rocket motor. Just prior to staging, when the booster engines are shut down and jet-tisoned, the control unit is automatically programmed (by command from the autopilot) to discontinue monitoring of the four booster engine switches.

d. Signals from a pair of differential pressure  $\Delta p$  sensors.

These switches monitor the difference in pressure across the intermediate bulkhead separating the two propellant tanks. Positive pressure (the fuel tank ullage pressure higher than the lox tank head pressure) is required to maintain rigidity of the bulkhead, and to assure structural integrity of the booster.

A pneumatic filter is placed in the sensing line between the liquid oxygen tank pressure port and the  $\Delta p$  sensors. This filter eliminates the effects of non-hazardous  $\Delta p$  transients which might otherwise trigger these switches into the abort condition.

e. Signals from three liquid oxygen (lox) tank ullage sensors.

Prior to staging, a relatively high level of pressure is required in this tank to maintain structural integrity under the heavy loads imposed by aerodynamic pressure. For this purpose, the control unit monitors only two of these sensors, those being set at the required high value. After staging, this requirement can be lowered; thus the control unit places the third sensor set to switch at a lower level, in parallel with the others. During this phase of flight, all three sensors must signal a dangerous pressure drop in order to initiate an abort signal from the control unit.

In addition to these sensor signals the control unit receives a-c and d-c power. The a-c line voltage is monitored by circuitry internal to the unit to maintain a constant check of the operation of the missile power system. The remaining major category of input signals are primarily programmer events. At lift off, the control unit is armed, enabling it to command an abort from that time on; at lift off plus 30 sec, the engine cutoff capability is enabled; at staging, signals are received to initiate proper modification of abort levels; and finally at sustainer cutoff, another signal disarms the system.

In case the vehicle should deviate from course, requiring a termination of the flight by the Range Safety Officer, the control unit receives a signal from the range safety system decoders. This signal instructs the system to shut down all ATLAS engines, initiate the abort sequence, and trigger a 3 sec timing device.

At the end of the 3 sec interval, the destruct package is armed and the Range Safety Officer may destroy the booster. This 3 sec interval is more than ample for the MERCURY capsule to be clear of the explosion.

The ASIS abort signal is generated by a drop out of voltage on two lines running to the capsule. The use of a zero signal to indicate abort rather than a positive voltage is chosen to enable the system to detect the loss of booster-capsule interface continuity. If a structural failure should cause the spacecraft adapter to fail, these wires would be broken, resulting in an immediate abort maneuver by the spacecraft. A similar set of wires is returned to the booster to provide assurance that the control unit is made aware of the spacecraft abort. This latter set of signals is also used in case the astronaut or flight director initiates an abort. In this case, the command is processed by the spacecraft itself and the control unit is notified through either a positive signal from the spacecraft or by loss of the interface continuity signals.

Detailed operation of the ASIS is beyond the scope of this paper. Further information is included in references (20) (21) (22) which describe the system in detail. Additional data may be obtained from the tabulation of ASIS parameters given in Table 2.\*

In-flight operation of the ASIS has been highlighted by several significant milestones. On July 29, 1960 the MERCURY/ATLAS - 1 mission (MA-1) was launched with the ASIS in an open loop configuration, i. e. all ASIS functions were telemetered to the ground but no capability for an actual abort command was included. The mission failed after approximately 50 sec of

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\* The information presented in Table 2 has been extracted from Report No. AE61-0474, The Project Mercury Abort Sensing and Implementation System, prepared by Convair (Astronautics) Division, General Dynamics Corporation, San Diego, California.

Table 2 ASIS Parameters

Parameter	Unit	Pre-staging Values		Staging Values		Post-staging Values	
		Value	Tolerance + -	Value	Tolerance + -	Value	Tolerance + -
Pitch Rates (Primary Set) (Backup Set)	deg/sec deg/sec	3.00 4.75	0.15 0.25	6.00 9.50	0.75 1.19	3.00 4.75	0.15 0.25
Yaw Rates (Primary Set) (Backup Set)	deg/sec deg/sec	3.00 4.75	0.15 0.25	6.00 9.50	0.60 0.95	3.00 4.75	0.15 0.25
Roll Rates (Primary Set) (Backup Set)	deg/sec deg/sec	6.40 9.40	0.30 0.40	6.40 9.40	0.30 0.40	6.40 9.40	0.30 0.40
LO <sub>2</sub> Tank Pressure	psig	19.0	1.00	11.00	1.00	11.00	1.00
LO <sub>2</sub> Fuel Tanks Differential Pressure	psid	2.5	1.00	2.5	1.00	2.5	1.00
Fuel Manifold Pressure, Booster one, Booster two	psia	470	25	none	-	none	-
Fuel Manifold Pressure, Sustainer	psia	560	25	560	25	560	25
Hydraulic Pressure, Sustainer	psia	2,000	60	2,000	60	2,000	60
Phase A, 400-cps Voltage	volts	70	10	70	10	70	10

## NOTES:

1. Pre-staging values are the parameter threshold values and tolerances from 2 in. motion to booster engine cutoff.
2. Staging values are the parameter threshold values and tolerances for 30 ± 0.5 sec following booster cutoff.
3. Post-staging values are the parameter threshold values and tolerances from booster engine cutoff plus 30 ± 0.5 sec to sustainer/vernier engine cutoff.
4. The over-rate detectors are monitored through filters having the following time constants:
  - A. Primary set, 225 ± 50 msec
  - B. Backup set, 80 ± 20 msec
5. The LO<sub>2</sub>/fuel tanks differential pressure is sensed through a pneumatic filter of 0.125 sec time constant.
6. Pitch, yaw, and roll rate values represent maximum allowable levels.  
Pressure and voltage values represent minimum allowable levels

flight. The ASIS signalled an abort through the telemetry link over 1 sec before the booster exploded. In this case, such a time interval would have been sufficient to assure spacecraft safety had the ASIS been in the closed loop, or active configuration. The recovered spacecraft wreckage, as shown by Fig. 13, was mute evidence of the importance of such a safety system. Had the system been in the closed loop configuration, in which it would have initiated capsule escape, the spacecraft would not have been damaged.

The next launch, MERCURY/ATLAS-2 (MA-2), on February 21, 1961 was the first closed loop ASIS mission. All booster systems performed normally and ASIS telemetry confirmed proper system response to all input stimuli, though no abort command was required.

The MA-3 mission on April 25, 1961 showed that the ASIS could generate an abort command if called upon to do so. A booster failure made it necessary for the Range Safety Officer to take action, resulting in an ASIS generated abort command to the spacecraft and proper arming of the destruct package. Although the booster was destroyed, the spacecraft was recovered unscathed, and was placed into orbit by the MA-4 booster. Since the MA-2 mission, all MERCURY/ATLAS boosters have operated with closed loop ASIS configurations. ASIS has performed perfectly in all cases.

Further details on the Mercury program may be obtained from references (23) through (26).

#### APOLLO Mission

Launch escape for the APOLLO mission will be implemented by jettison of the command module as shown in Fig. 14. The high impulse escape tower to be used will incorporate a three nozzle configuration motor having thrust on the order of 200,000 lb. Removal of the tower from the command module will be effected by operation of a tower jettison rocket which will have approximately

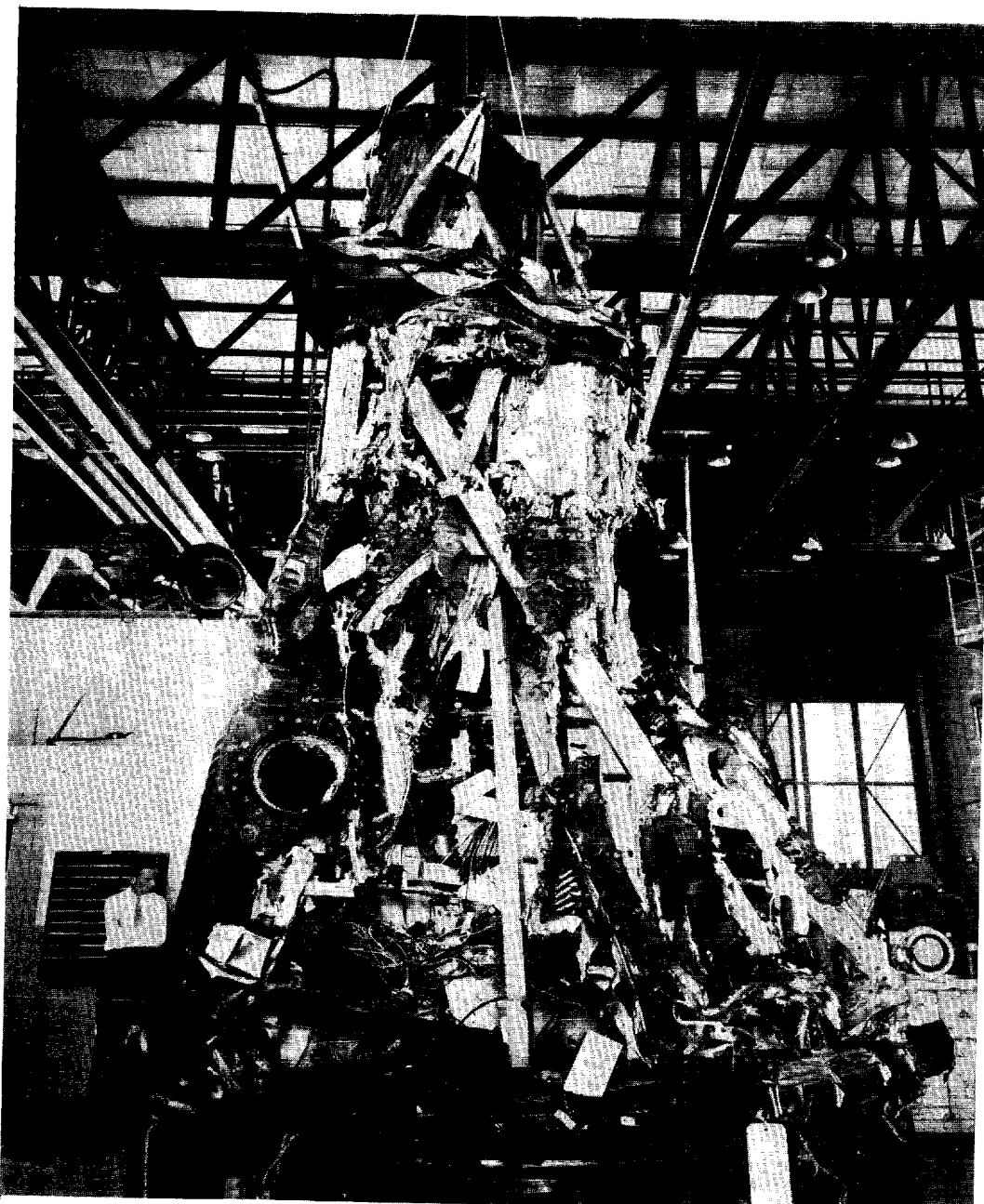


Fig. 13 MERCURY/ATLAS - 1 capsule after post flight  
salvage and reassembly

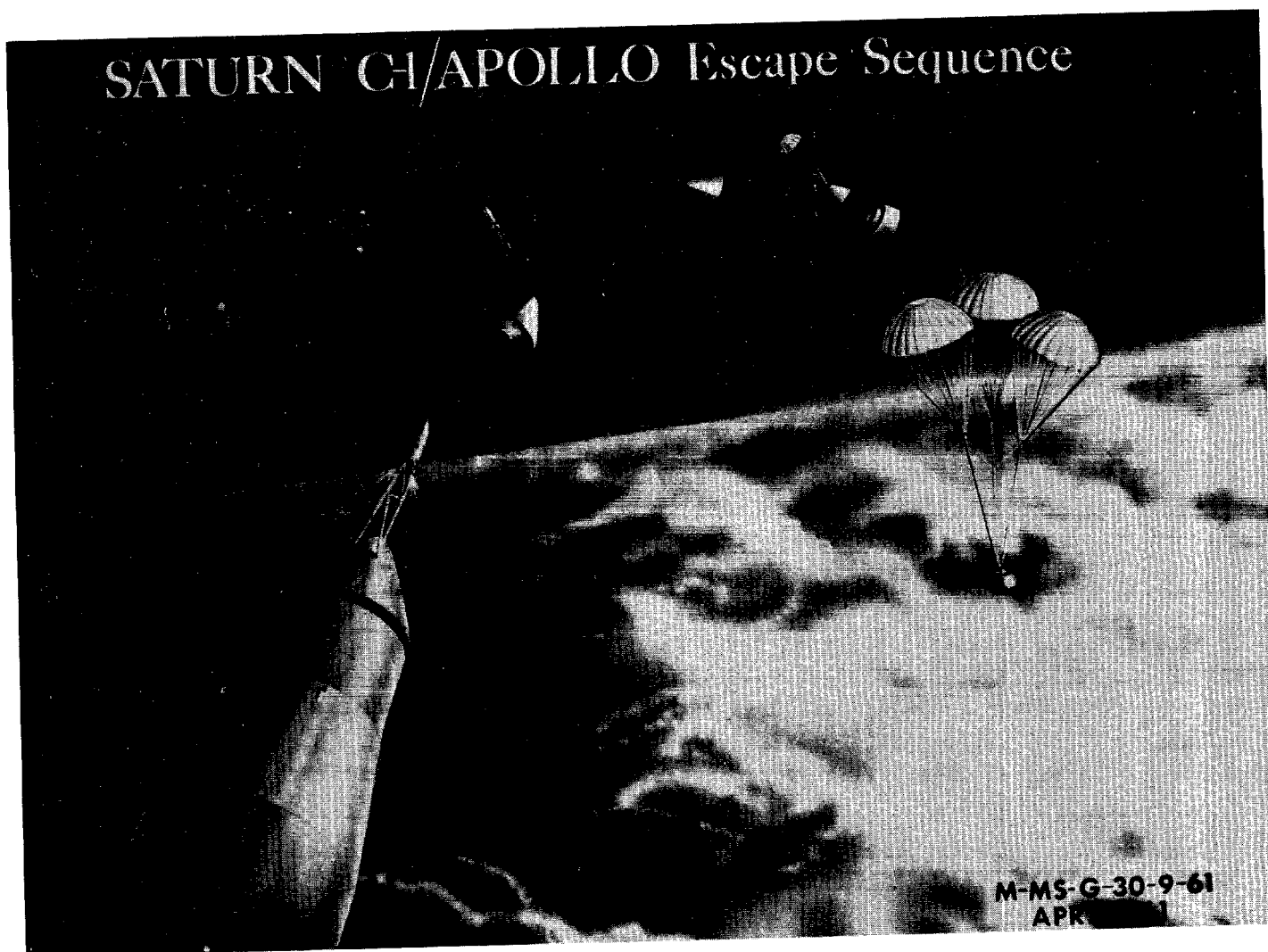


Fig. 14 Artist's conception of an APOLLO abort

40,000 pounds thrust. In the event of an abort after the tower has been jettisoned, either during booster phase or during the trans-earth coasting phase abort propulsion will be provided by the Service module propulsion unit. This unit will depend upon a single combustion chamber developing approximately 20,000 lb thrust, fed by a liquid hypergolic mixture of nitrogen tetroxide and a fifty-fifty mixture of unsymmetrical dimethyl hydrazine and hydrazine. The APOLLO abort sensing system, to be designed for the SATURN and ADVANCED SATURN boosters is still in extremely preliminary form, and details are therefore not available at this time. The SATURN which is to be used for the early APOLLO orbital missions is shown in Fig. 15.

#### DYNA SOAR Mission

The DYNA SOAR winged glider is to be placed into orbit by the TITAN III booster. This booster is a combination of the liquid propellant TITAN II to be described below, and two solid fuel segments to provide the necessary additional capability. The abort system for both the TITAN II and the TITAN III is known as the MALFUNCTION DETECTION SYSTEM<sup>(27)</sup> and will operate on principles similar to those used in the REDSTONE system, though the system will be much more complicated. Abort will be initiated by firing a rocket within the DYNA SOAR craft itself, similar to the procedure shown in Fig. 16. Information on the techniques which may be used for escape from a winged spacecraft in case the spacecraft itself must be abandoned may be found in reference (28).

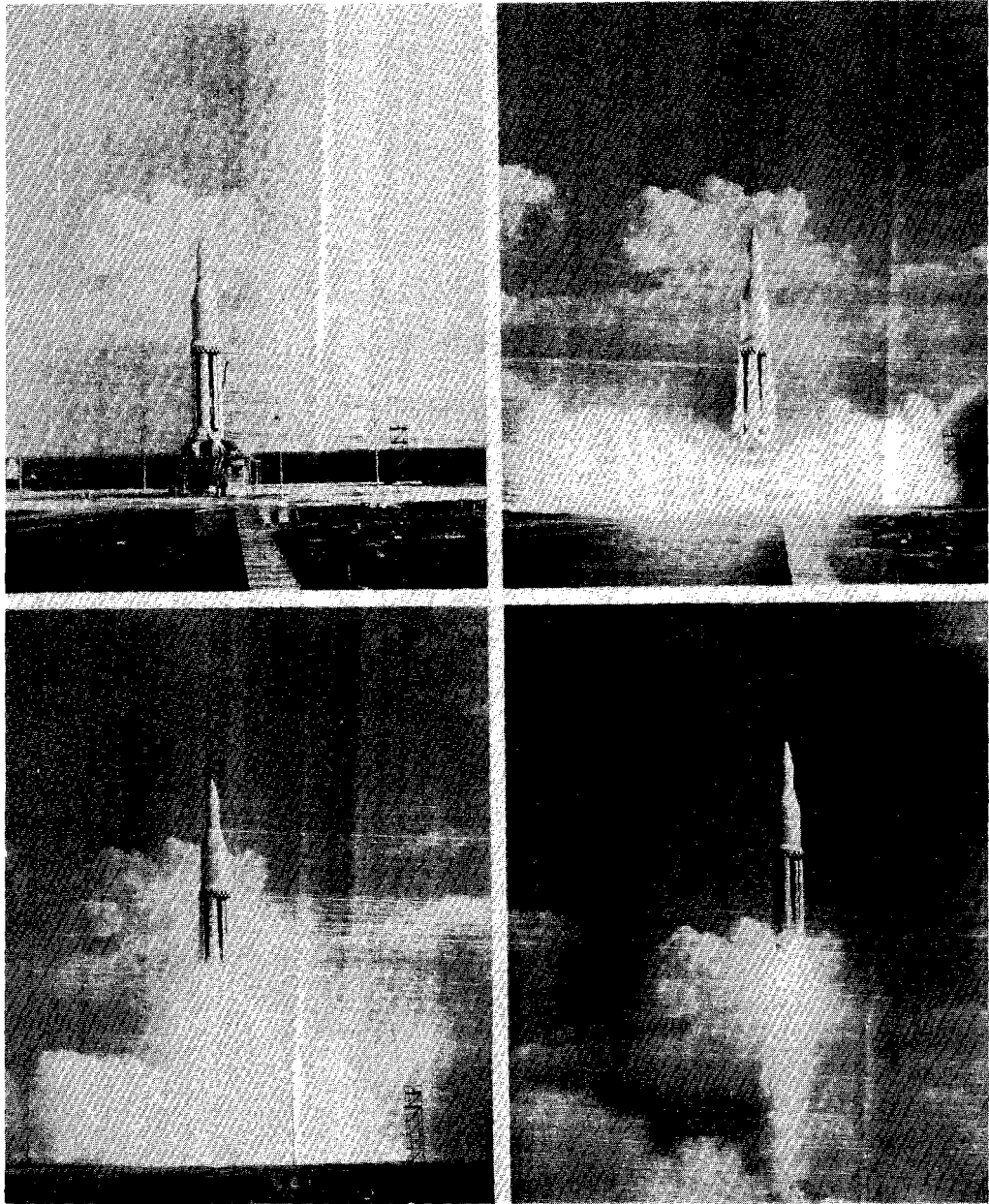


Fig. 15 SATURN launch



Fig. 16 Artist's conception of DYNA-SOAR type vehicle  
in abort maneuver

### GEMINI Mission

The GEMINI spacecraft is to be put into orbit by a TITAN II similar to that shown in Fig. 17. <sup>(29)</sup> In the GEMINI configuration, the vehicle will stand 107 ft high and 10 ft in diameter. Using hypergolic storable propellents the first stage will deliver 430,000 lb thrust, while the second stage is rated at 100,000 lb thrust. The external configuration of this two man craft is seen to be similar to the MERCURY capsule (Fig. 18). This configuration does not incorporate the familiar escape tower. Abort is implemented through twin ejection seats designed to catapult the men from the vehicle. The use of such a technique, particularly at extremely high altitudes, requires the solution of additional problems associated with such a descent through the upper atmosphere. These problems are caused by two factors. First, the hostile environment with which the individual is faced in case of an abort at an altitude above 70,000 ft. Second, the high velocity of his body at ejection would make it extremely hazardous to open his parachute at this high altitude due to the opening shock, which could easily exceed 40 g's. It is imperative, therefore, that he initiate a free fall which must last until his velocity has decreased close to the lower atmosphere terminal velocity at 120 mph in order to prevent this high g loading upon parachute deployment. Such action is also necessary to remove him from the ultra low pressure and temperature environment as quickly as possible.

The initiation of a rapid descent, while necessary for the above reasons, has associated with it a very serious and little known danger. This is the so-called flat spin, a rotation about the pitch (front to back) axis caused by aerodynamic forces exerted upon the rapidly falling body. Early experiments with anthropomorphic dummies dropped from large polyethylene balloons at heights of up to 91,000 ft indicated rotational rates reached 200 rpm in some cases. Velocity of fall reached over 700 mph during

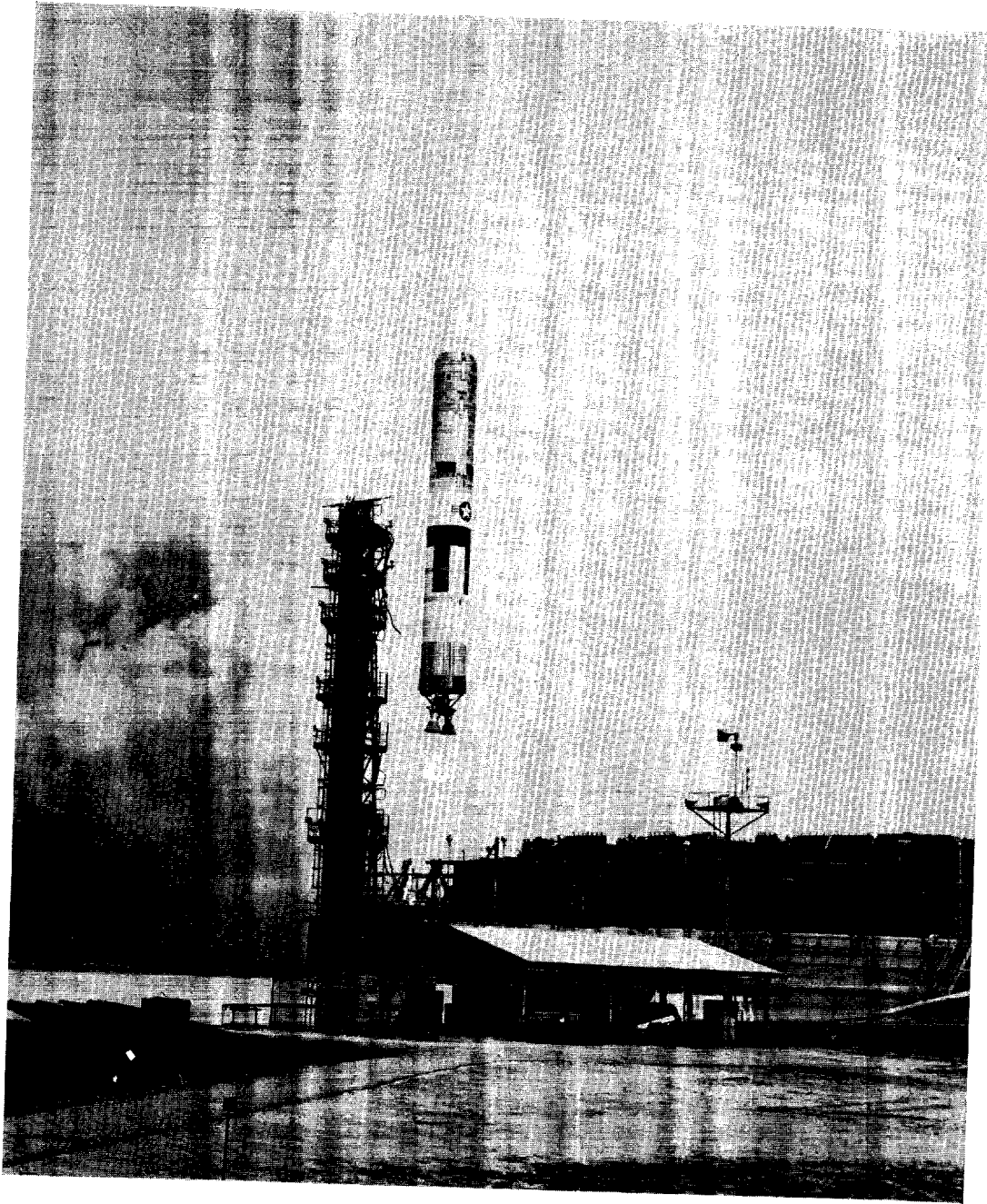


Fig. 17 TITAN II launch

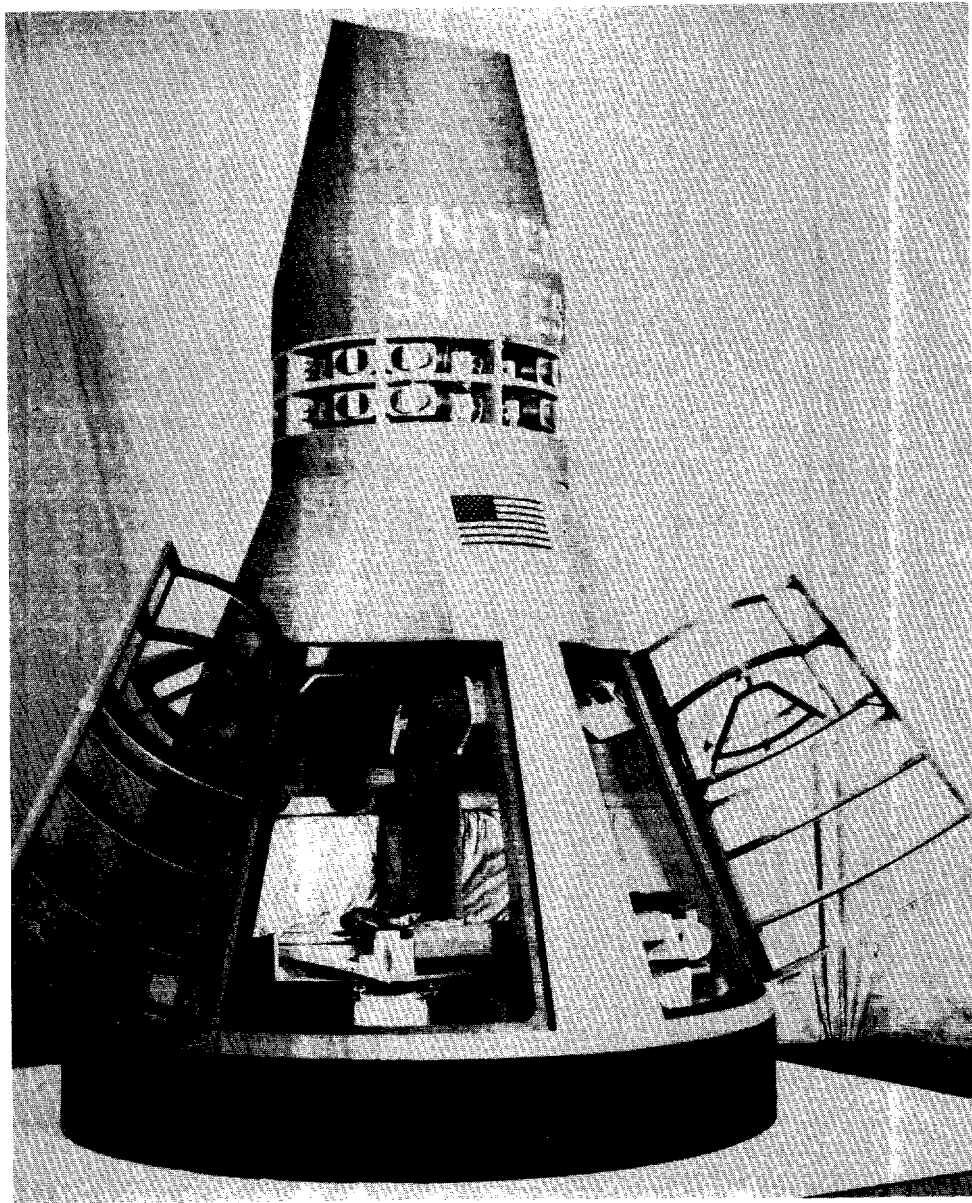


Fig. 18 GEMINI - two man spacecraft

the high altitude portions of the descent. Further experiments were conducted under the code names of High Dive, Man High, and Excelsior, including a number of manned descents. The manned jumps were made on November 16 1959, December 11, 1959, and August 16, 1960, from altitudes of 76,000 ft, 74,400 ft, and 102,800 ft, respectively, by Captain Joseph Kittinger, USAF. (See Fig. 19) The jumps, although initially marred by near disaster, demonstrated the possibility of making semi-free falls through use of a specially designed stabilization parachute deployed at a high altitude. This device does not significantly reduce the rate of descent, but it does provide proper stabilization until deployment of the main parachute (at altitudes of less than 20,000 ft). Further details of these missions and the associated problems are to be found in reference (30).

Much data has also been obtained from the design of escape systems of a number of contemporary high performance aircraft. The use of protective suits and automatic descent sequence (seat separation, parachute deployment etc) to safeguard the crew was developed in conjunction with these programs.

For aborts above 70,000 feet the spacecraft is separated from the booster, makes a normal re-entry, and using an inflatable paraglider (Rogallo wing) makes a controlled landing. The crew may also eject from the spacecraft after a suitable low altitude is reached, if they so desire.

#### VOSTOK Mission

Recovery of the cosmonauts from flights aboard VOSTOK class vehicles is conducted by either of two methods: landing within the spacecraft as it is lowered by parachute, or by individual parachute. In the latter case, ejection from the spacecraft is accomplished through use of an ejection seat (Fig. 20\*) activated

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\*An adaptation of SOVFOTO from Gherman Titov and Martin Caidin's I Am Eagle (Published by the Bobbs-Merrill Company, Inc. New York).



Fig. 19 Start of jump made by Captain Joseph Kittinger  
from 102,800 ft, August 16, 1960

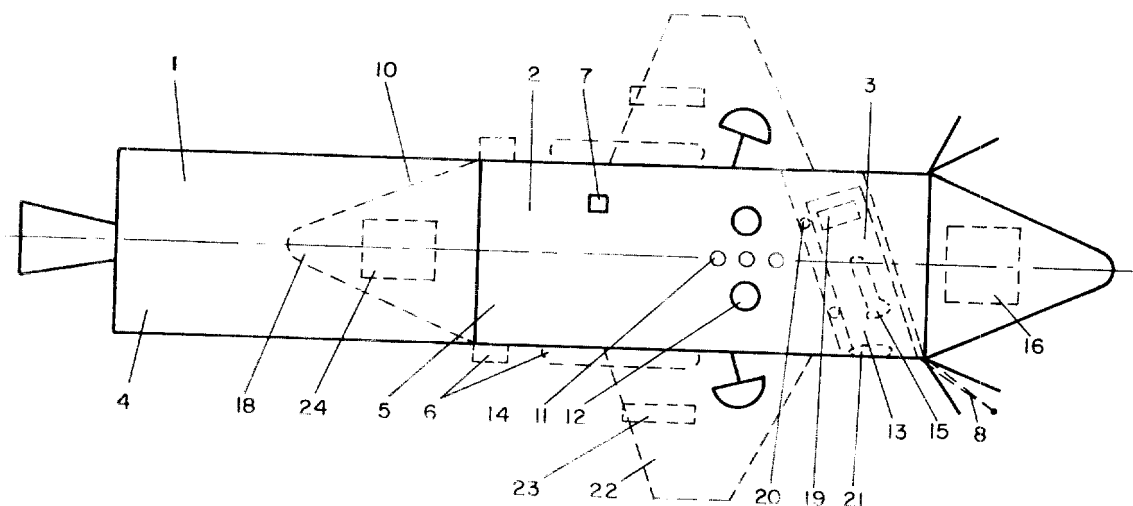


Fig. 20 Ejection seat - VOSTOK spacecraft

prior to the deployment of the spacecraft internal parachute system. This seat exit is also employed in case of abort during launch phase. Major Titov, pilot of the VOSTOK II spacecraft, as well as cosmonauts Nikolayev and Popovich, have stated that they landed safely in this manner near the recovery site at Krasney Koot. Extensive tests of this system were conducted on animals, culminating in the successful ejection of the dogs Strelka and Belka and a number of smaller animals from the second ship satellite launched August 19, 1960. The catapultable container, containing the animals was located within the recoverable cabin section of the ship. Landing speed of the container was 7 meters/sec; of the cabin itself, 10 meters/sec, indicating that safe landings may be made within the spacecraft. The second ship satellite, a configuration quite similar to the VOSTOK vehicles is illustrated in Fig. 21. Note the instrument compartment containing the retro-thrust propulsion plant which separates from the cabin prior to re-entry and is not recovered.

Preparation for space flight by the twelve Soviet Cosmonaut trainees laid particular emphasis on the landing technique. Each is required to make over 40 parachute jumps into varied types of terrain prior to flight qualification.

It may be safely assumed that the two stage booster rocket (1,300,000 pounds total thrust) which launches the VOSTOK vehicles from the Baikonur cosmodrome carries a malfunction detection system similar in principle to those described previously. For further details of the VOSTOK program, the reader is referred to the bibliography, references (31) through (39).



Legend: 1-nonrecoverable instrument section; 2-recoverable cabin; 3-catapultable container; 4-equipment in instrument section: radiotelemetric apparatus, apparatus for controlling flight of ship-satellite, part of apparatus for scientific investigations (instruments for study of cosmic rays and short-wave solar radiation), apparatus of heat regulation, and deceleration power plant; 5-equipment in cabin: devices to assure normal functioning of the animals in the flight, equipment for biological experiments, part of apparatus of orientation system, apparatus for registration of animals' behavior in cabin during re-entry (pickups indicate angular velocity, overloads, temperatures, noises, etc.), automatic systems for landing the ship, apparatus for autonomic registration of instrument data and of physiological data of tested animals during deceleration; 6 to 10-equipment located on the external surface of the cabin including 6-rudder nozzles and tanks with compressed gas of the control system, 7-pickups of scientific apparatus, 8-an-tennas of the radio systems, 9 self-orienting experimental sun batteries in the form of two half-disks with diameter of 1 m, 10-system of thermoinsulation for protection of cabin during deceleration; 11 and 12-equipment located in cabin walls, including 11-heat-resisting observation parts and 12-quick-opened hermetic manholes; 13-equipment in container: animal compartment with cradle, automatic feeding device, sanitary device, ventilation system, etc., catapulting and pyro-technical equipment, direction-finding radio transmitters container, television cameras with system of illumination and mirrors, and blocks with nuclear photo-emulsions; 14-experimental subjects in cabin: 28 laboratory mice and 2 white rats; 15-experimental subjects in container: 2 dogs, 12 mice, insects, plants, mushroom cultures, corn, wheat, pea, and onion seeds, certain species of microbes and other biological objects [128]; 16 to 24-possible equipment used in the second ship-satellite, including 16-fabric parachute for cabin, 18-re-entry cone, 19-fabric parachute for container, 20-rollers of catapulting system, 21-hatch for catapulting container, 22-cabin wings, 23-deceleration engines of cabin, and 24-

Fig. 21 Possible structure of second Soviet ship-satellite (SPUTNIK V)

## CHAPTER 4

### CONCLUSIONS

A number of advanced methods are under consideration for the return of astronauts and spacecraft from space. A particularly interesting method, which could have significant application to both normal missions and abort situations, is the use of the Rogallo wing. <sup>(40)</sup> This device, taken from a design originally conceived by Leonardo Da Vinci, is currently the subject of a number of experiments. The flex wing aircraft shown in Fig. 22, a manned test bed for experiments concerned with the determination of the wing's performance characteristics, is one such experiment. A 180 horsepower engine drives the rearward-pointing propeller supplying power to the craft. Lift is supplied by the forty foot wing structure which supports the 1,500 lb structure.

Another such escape and landing system makes use of a set of extendable rotary wings. Tip jets fed by blade tanks provide power for the final letdown maneuver. <sup>(41)</sup>

As the space programs of the future progress into multi manned extended missions, the problems of crew safety will become even more acute. The many current configurations of launch and orbital vehicles will be joined by permanent space stations, space ferries, interplanetary carriers, and many other craft which will carry man through the space environment. The design and development of appropriate failure monitoring equipment and apparatus to assure the safe rescue or the return to earth of space crews will be a significant challenge for some time to come. These devices will be necessary to raise the level of safety of space voyagers to that now enjoyed by the pilots of modern high performance aircraft.

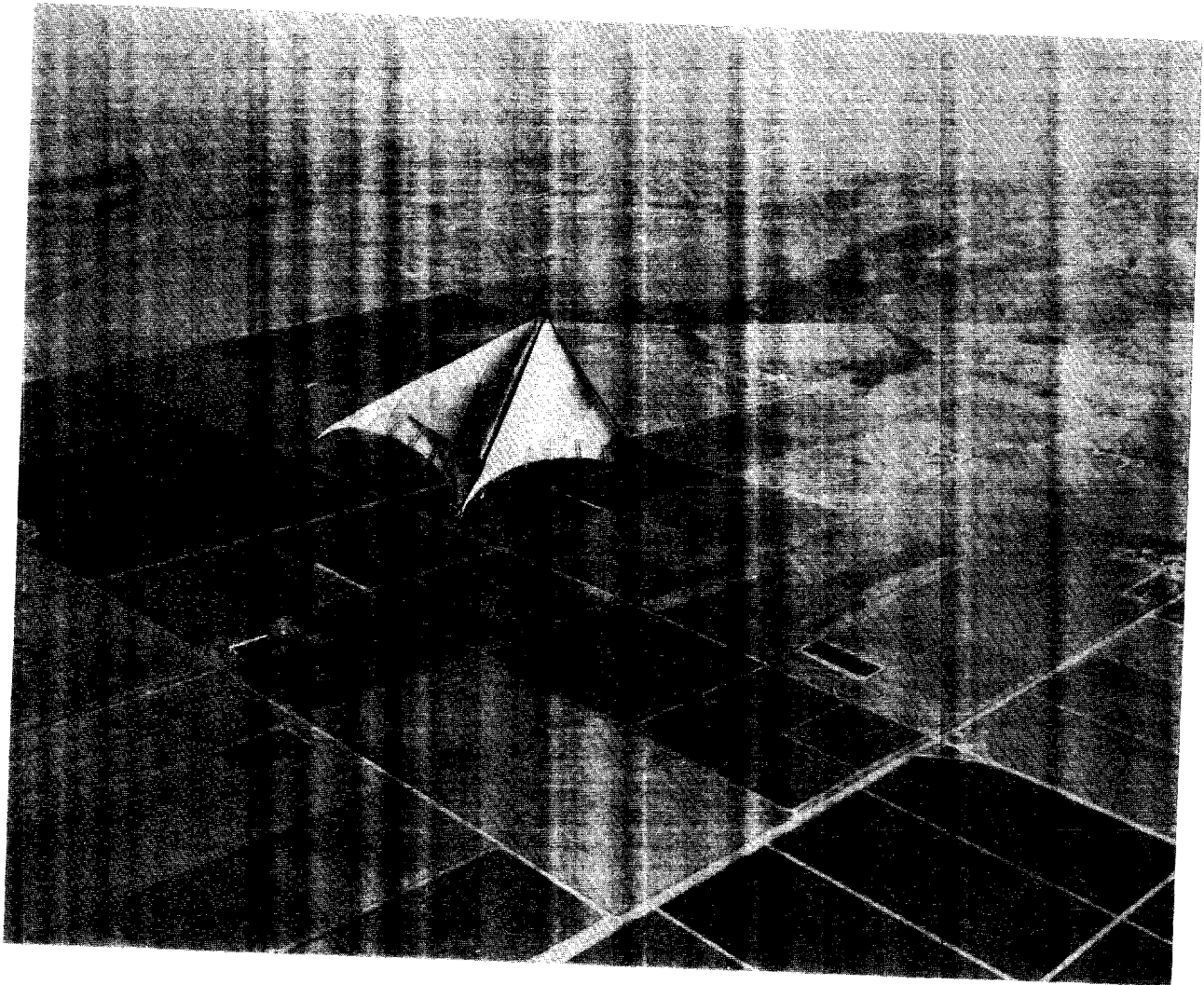


Fig. 22 Flex wing in-flight

BIBLIOGRAPHY

1. Kelly, T. J. and Adornato, R. J., Determination of Abort Way-Stations on a Nominal Circumlunar Trajectory, American Rocket Society Report 2085-61, October 1961.
2. Hakes, R. C., Guidance and Navigation Systems for Lunar Missions, American Rocket Society Report 2294-61, October 1961.
3. Miller, J. S. and Deyst, J. J., Preliminary Study of Aborts From Circumlunar Trajectories, Report E-1124 (Instrumentation Laboratory, Massachusetts Institute of Technology, March 1962).
4. Results of the First United States Manned Orbital Space Flight, National Aeronautics and Space Administration, Manned Spacecraft Center, March 1962.
5. Hessberg, R. R. Lt. Col. USAF (MC), Space Flight Dynamics, (Aeromedical Field Laboratory, Holloman AFB, New Mexico, 12 January 1960).
6. Stapp, J. P. Col. USAF (MC), "The Effects of Mechanical Force on Living Tissue," Journal of Aviation Medicine, August and October 1956.
7. "The Human Factor in Space Travel," United States Air Force Air University Quarterly Review, Vol. X, No. 2, Summer 1958.
8. United States Air Force Manual AFM 160-30, Physiology of Flight (Department of Air Force).
9. Fisher, A. C. Jr., "Aviation Medicine on the Threshold of Space," National Geographic Magazine, Vol. 108, No. 8, August 1955.
10. Strughold, H., Space Medicine and Astrobiology, presented to the XI International Astronautical Federation Congress, Stockholm, Sweden, 15-20 August 1960.
11. Life Sciences and Space, 86th Congress, 2nd Session, House of Representatives Report No. 2227.
12. Experimental Track Branch (Air Force Flight Test Center, California, 1961).

13. Gazenko, O.G. and Yazdovsky, V.J., Some Results of Physiological Reactions to Space Flight Conditions, presented to XII International Astronautical Federation Congress, Washington D.C., 4 October 1961.
14. Boyce, W.C. and Freeman, H.E., Considerations Affecting the Design of a 60 G Personnel Restraint System, American Rocket Society Report 2105-61, October 1961.
15. Woodson, W.F., Human Engineering Guide for Equipment Designers (University of California Press, Berkeley: 1956).
16. Kuettner, J.P., "Rocket Reliability," Spaceflight, British Interplanetary Society, Vol. III, No. 5, September 1961.
17. Results of the First U.S. Manned Suborbital Space Flight, National Aeronautics and Space Administration, Manned Spacecraft Center, 6 June 1961.
18. Mercury - Atlas MA-7 Launch Information Notebook, General Dynamics/Astronautics, San Diego, California, May 1962.
19. Special Report No. 62-27 (Mercury - Atlas Booster), Office of the Deputy Commander, Air Force Systems Command, for Aerospace Systems, 1962.
20. Gadbois, R.D., Heinsheimer, T.F., and Schaelchlin, J.W., "Atlas Missile Pilot Safety System for Project Mercury," IRE Convention Record, Space Electronics Session, 1961. (An abbreviated form of this report is "Automatic Abort for Mercury - Atlas," Missiles and Rockets, 3 April 1961.)
21. Gadbois, R.D., Heinsheimer, T.F., and Schaelchlin, J.W., "Mag Amp Monitors Mercury - Atlas Gyros," Control Engineering, December 1961.
22. Abort Sensing and Implementation System for Project Mercury, General Dynamics/Astronautics Reprot #AE-61-0474, San Diego, California, 21 May 1961.
23. Results of the Second United States Manned Orbital Space Flight, National Aeronautics and Space Administration and Spacecraft Center, May 24, 1962.
24. Rose, Rodney G. and Armitage, Peter J., Project Mercury Water Landing Problems, presented to I.A.S. 30th Annual Meeting January 22 - 24, 1962.
25. Jones, Edward R., "The Training of Astronauts", Man's Integration into the Mercury Capsule, National Research Council, Washington, D.C. 1961; pages 15 - 21.

26. Holmes, D. B., Statement made before the Subcommittee No. 1, Committee on Science and Astronautics, House of Representatives, Spring 1962, National Aeronautics and Space Administration Document.
27. "Industry Observer," Aviation Week and Space Technology, 9 April 1962.
28. Garnett, R. J., The Selection of an Escape System for a Boost Glide or Orbital Re-Entry Vehicle, American Rocket Society Report 2213-61, October 1961.
29. News Release No. REL 61-41 (Titan II Fact Sheet), Office of the Deputy Commander, Air Force Systems Command, for Aerospace Systems, 1961.
30. Kittinger, J. W. Captain USAF, The Long, Lonely Leap (E. P. Dutton & Co., Inc., New York: 1961).
31. Titov, G. S. Major, USSR Air Force, I Am Eagle! (Bobbs - Merrill, New York: May 1962)
32. "Special Report on Flight of Yuri A. Gagarin, Major, USSR Air Force," Soviet Union, Issue No. 136, May 1961.
33. The First Man in Space, USSR Government Publication, May 1961.
34. Caidin, M., Red Star in Space (Collier Books, to be published on 28 October 1962).
35. Behind the Sputniks: A Survey of Soviet Space Science, United States Air Force Project RAND Report R-311, 3 November 1957.
36. A Casebook on Soviet Astronautics, United States Air Force RAND Research Memorandum RM-1760, 21 June 1956.
37. A Casebook on Soviet Astronautics - Part II, United States Air Force RAND Research Memorandum RM-1922, 21 June 1957.
38. "Designing the Vostok," Spaceflight, British Interplanetary Society, Vol. III, No. 5, September 1961.
39. Comprehensive Analysis of Soviet Space Program, AID Report 61-72, 22 May 1961.

40. "I Fly the Flex Wing . . . , " Ryan Reporter, Vol. 22, No. 2, Ryan Aircraft Corp. , San Diego, California, 1962.
41. Hamilton, J. S. , Large Booster Recovery Techniques, American Rocket Society Report No. 2049-61, October 1961.